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SPACE OPERATIONS CENTER — SHUTTLE INTERACTION STUDY (NAS9-16153)

SSD-81-0076

FINAL REPORT

VOLUME I

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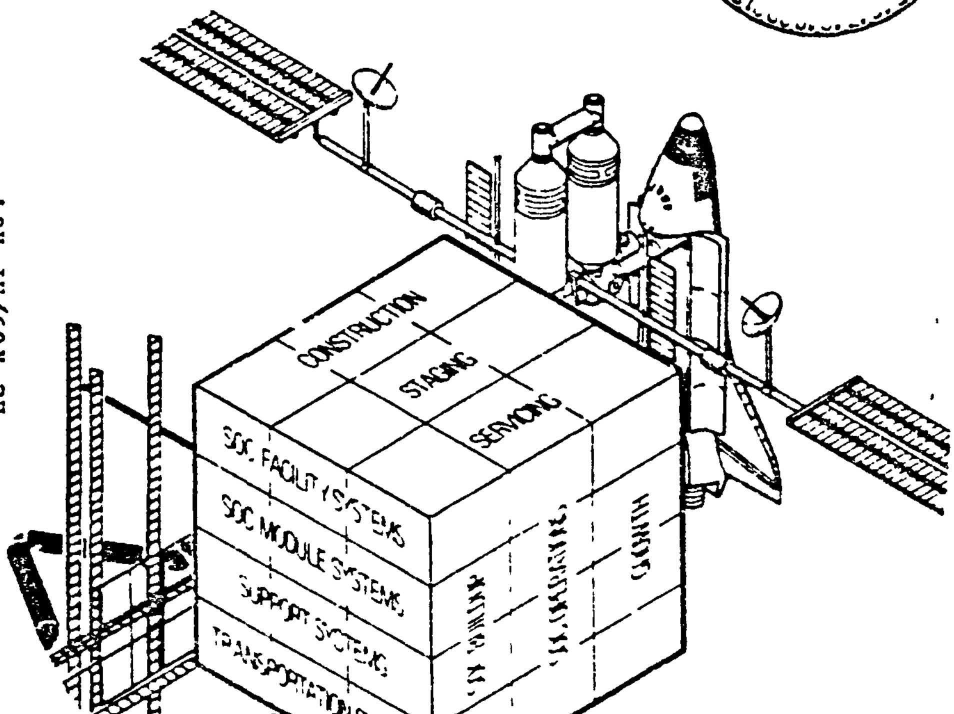

DRL T-1626

LINE ITEM 3

April 17, 1981

Space Operations and Satellite Systems Division

Rockwell International

N83-13141 #

SSD 81-0076

SPACE OPERATIONS CENTER/SHUTTLE INTERACTION STUDY

FINAL REPORT, VOLUME I

Contract No. NAS9-16153
DRL T-1626
Line Item 3

April 17, 1981

Approved by a J Stefan
A. J. Stefan

Rockwell International

Space Operations and
Satellite Systems Division

FOREWORD

This report contains the results of the analysis that was performed to determine the implication of using the Shuttle with the Space Operations Center (SOC).

This effort was performed under Contract Number NAS9-16153, by the Space Operations and Satellite Systems Division of Rockwell International for the National Aeronautics and Space Administration, Johnson Space Center. The study was administered under the technical direction of the Contracting Officers Representative (COR), Mr. S. H. Nassiff, Program Development Office, Engineering and Development Directorate, Johnson Space Center.

The study was performed under the direction of Ellis Katz and A. J. Stefan, Study Managers. The following persons made significant contributions to the completion of the analysis.

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INTRODUCTION

The Space Operations Center (SOC) is conceived as a permanent facility in low earth orbit incorporating capabilities for space systems construction; space vehicle assembly, launching, recovery and servicing; and the servicing of co-orbiting satellites.

The Shuttle Transportation System (STS) is an integral element of the SOC concept. It will transport the various elements of the SOC into space and support the assembly operation. Subsequently, it will regularly service the SOC with crew rotations, crew supplies, construction materials, construction equipment and components, space vehicle elements, and propellants and spare parts.

This report contains the results of the study that analyzed, in a preliminary fashion, the implication of using the Shuttle with the SOC, including constraints that the Shuttle will place upon the SOC design. The study identifies the considerations involved in the use of the Shuttle as a part of the SOC concept, and also identifies the constraints to the SOC imposed by the Shuttle in its interactions with the SOC, and on the design or technical solutions which allow satisfactory accomplishment of the interactions.

STUDY TASKS

Five specific task areas were identified for study. These tasks are indicated in Table I-1 along with the principal issues associated with each task.

TABLE I-1 STUDY TASKS

TASK	ISSUES
1.0 ORBITAL ALTITUDE	• AT WHAT ALTITUDE SHOULD THE SOC OPERATE WHILE BEING COMPATIBLE WITH THE SHUTTLE CAPABILITIES?
2.0 BERTHING AND/OR DOCKING	• IS A STANDARD BERTHING/DOCKING INTERFACE FEASIBLE? • CAN THE ORBITER DOCK TO THE SOC? • CAN THE ORBITER BERTH TO THE SOC USING THE RMS?
3.0 SOC ASSEMBLY	• WHAT EQUIPMENT AND OPERATIONS ARE REQUIRED FOR THE SHUTTLE TO ASSEMBLE THE SOC? • WHAT ARE THE IMPLICATIONS TO THE SOC ELEMENTS?
4.0 SOC RESUPPLY AND FUEL TRANSFER	• WHAT ARE THE IMPLICATIONS OF SOC RESUPPLY VIA THE LOGISTICS MODULE AND THE SHUTTLE? • WHAT ARE THE IMPLICATIONS OF TRANSFERRING PROPELLANTS FROM THE SHUTTLE TO THE SOC? • DEVELOP A SHUTTLE LOGISTICS MODEL
5.0 FLIGHT SUPPORT FACILITY	• WHAT ARE THE IMPLICATIONS TO THE SOC TO PROVIDE SPACECRAFT SERVICING? • WHAT ARE THE IMPLICATIONS TO THE SHUTTLE TO PROVIDE SPACECRAFT SERVICING? • WHAT ARE THE SPACE-BASED VEHICLE REQUIREMENTS?

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SPACECRAFT CONFIGURATIONS

The reference SOC configuration supplied by NASA/JSC was utilized as the model for this study, Figure I-1. Changes to the configuration were only imposed when the task implications so indicated. Two OTV concepts were utilized as models for the analysis, a parallel stage arrangement that was developed from the manned OTV study by Grumman Aerospace Corporation for NASA/JSC, Figure I-2, and a tandem stage concept, Figure I-3, developed from the Future OTV Technology Study by Boeing Aerospace Corporation for NASA/LARC. Basic configuration and operational characteristics of the OTV stages were mutually developed with Rockwell International, Boeing Aerospace Corporation, and NASA/JSC. These characteristics are listed in Section 5.0 Flight Support Facility.

REPORT ORGANIZATION

This report is organized into five basic sections that correspond to the five tasks previously described. A conclusion section will summarize the implications to the SOC and to the orbiter, and will describe the requirements imposed on the OTV as a result of the space based servicing operations.

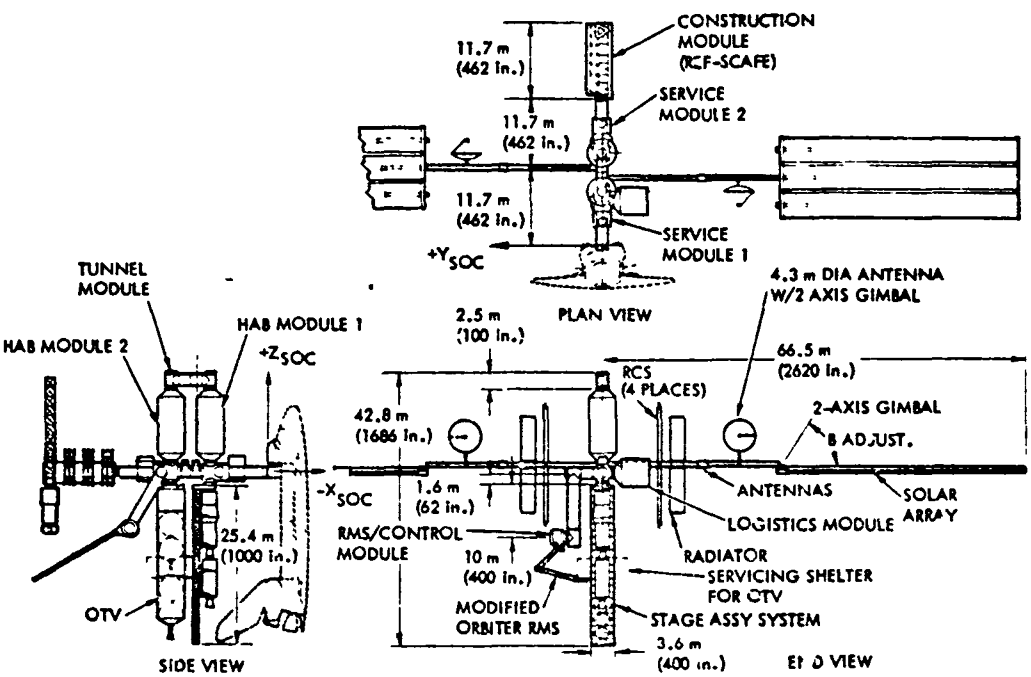

FIGURE I-1 SPACE OPERATIONS CENTER REFERENCE CONFIGURATION

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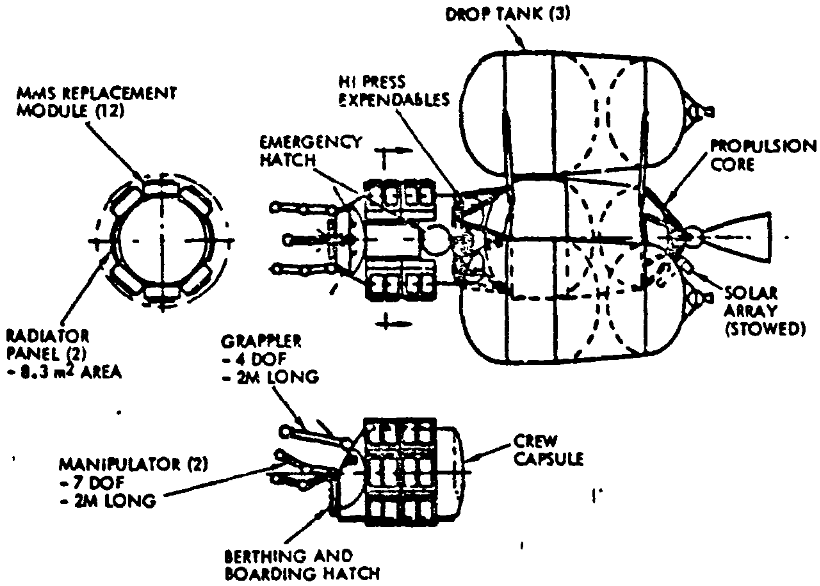

FIGURE I-2 MOTV GEO TRANSFER CONFIGURATION

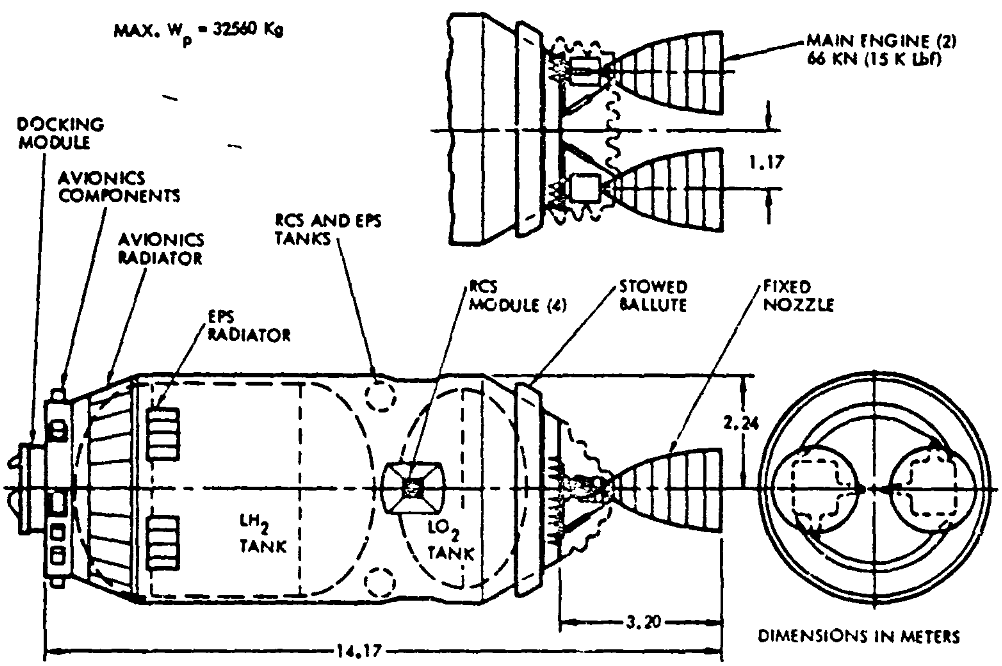

FIGURE I-3 SPACE-BASED OTV CONFIGURATION

1.0 ORBITAL ALTITUDE

1.1 SUMMARY

The Space Operations Center (SOC) is planned as a versatile permanently manned base in low earth orbit. The space shuttle will resupply the SOC on a regular basis to support a wide variety of missions. Orbit transfer vehicles (OTV) will be refueled and serviced at the SOC to carry payloads on missions to higher energy orbits and large space systems, too large for single flight delivery in the orbiter bay, will be erected, assembled and/or constructed on special SOC facilities.

The size, complexity and value of the Space Operations Center and the associated space systems requires that a careful and detailed analysis be performed to determine preferred operating altitudes. The objective of Task 1.0, then, is to seek out the most effective orbit altitude strategy for the SOC which utilizes the maximum potential of the Space Shuttle and at the same time provides adequate safety and an efficient operating base for staging OTV missions.

There are a number of factors which can influence the selection of SOC orbit altitude. The most prominent of these are shown symbollically in Figure 1.1. The nature of their effects are highlighted in the following discussion. First, the delivery system performance for SOC resupply is of obvious importance. Payload performance drops off with altitude for all launch systems and affects the number of flights required to deliver a given amount of cargo. Thus, the specific performance characteristics of the standard shuttle and later thrust augmented versions as well as other logistics modes such as tug assisted concepts can affect the desired SOC orbit altitude. Generally, the shallower the slope of payload drop off with altitude the higher the preferred operating orbit will be. This is because at the lower altitudes where payload delivery performance is high, the drag forces acting on the SOC are also high, thereby requiring large quantities of orbit makeup propellant.

The actual amount of required propellant and hence its importance in altitude determination is influenced by several factors. They are atmospheric density, which varies dramatically over the 11 year solar cycle period, the drag configuration of the SOC, which can also vary depending upon construction project and OTV activity levels, and the SOC propulsion system specific impulse. The balance between these two basic effects, delivery performance drop off at high altitudes and high drag makeup propellants at low altitudes determine the optimum logistics altitude requiring the least number of delivery flights.

However, these factors are further influenced by the amount and nature of the SOC logistics traffic. High traffic levels mean the Shuttle delivery performance begins to outweigh the orbit makeup propellant effects on optimum altitude, and for high traffic levels the optimum altitude will be lower. The actual altitude will also be influenced by the packaged density

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characteristics of the logistics cargo. Volume limited cargo can be carried to higher orbit altitudes, thus putting a "bias" in the delivery performance effects on the optimum orbit altitude.

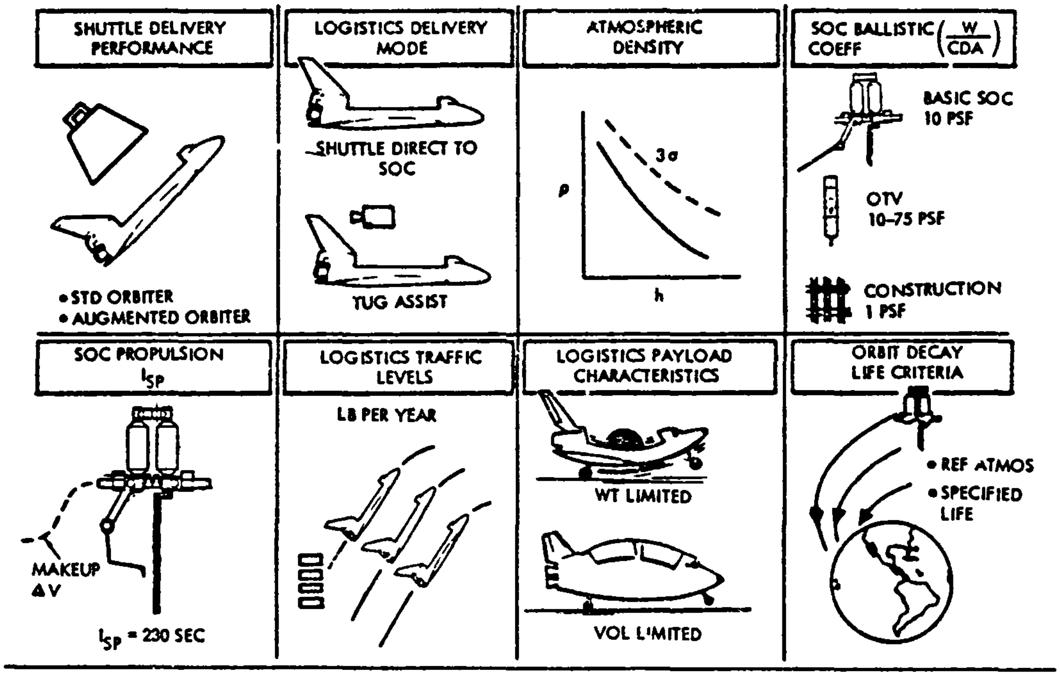

FIGURE 1.1 SOC ORBIT ALTITUDE FACTORS

Superimposed on all of the above factors is the need for orbital safety. A specified orbit decay life criteria will set altitude limits, which at times, depending upon the actual criteria selected and the prevailing SOC logistics environment, may be the governing condition.

The analysis of these factors and their interactions resulted in the following principal conclusions which are highlighted below along with brief substantiating text.

A Variable Altitude Strategy is Recommended for SOC Operation

A variable altitude strategy as depicted in Figure 1.2 combines safety with logistics efficiency. During periods of unusually high solar activity the SOC orbit altitude would be adjusted upward to maintain the 90-day orbit decay life criteria required for orbital safety. However, most of the time, when solar activity levels follow their nominal 11 year cycle trends, the SOC altitude can be greatly reduced to take advantage of the greater shuttle payload delivery capability at low altitudes. This improves the logistics efficiency by reducing the number of shuttle flights required to deliver a given amount of SOC cargo. Further, the actual operating altitude can be optimized for the prevailing atmospheric density and amount of SOC logistics traffic scheduled. This variable altitude approach can save 10 to 15 percent in the number of required shuttle flights to SOC compared to a constant altitude concept which must be based on the worst case decay environment and hence must always fly at a high altitude. Thus, a variable altitude strategy is recommended.

The Standard Orbiter can do the Job

The currently projected modular elements of the SOC configuration, such as the service modules, the habitability modules, etc., can all be delivered to orbit by the standard shuttle. These various modules, logically sized for their respective SOC mission roles, fit within the orbiter cargo bay and are well within the payload delivery capability of the standard shuttle. Normal SOC resupply, OTV propellants and other SOC cargo can also be delivered by the standard shuttle.

The extra payload capability of the thrust augmented shuttle is not needed for the delivery of the SOC modules. However, if cost effective in terms of dollars per pound to orbit, it may prove to be more efficient for OTV propellant deliveries, but even here the standard shuttle is sufficient. The optimum SOC altitude is about 18 Km (10 nmi) higher with the augmented thrust shuttle, but varies with logistics traffic levels and density in the same manner as the standard shuttle. Therefore, both the standard and augmented shuttles are compatible with the variable altitude strategy.

Thus, while gains in logistics efficiency for weight limited payloads such as OTV propellant deliveries may be attainable with the thrust augmented shuttle the standard shuttle can do an adequate job. A special new delivery system is not required for the SOC.

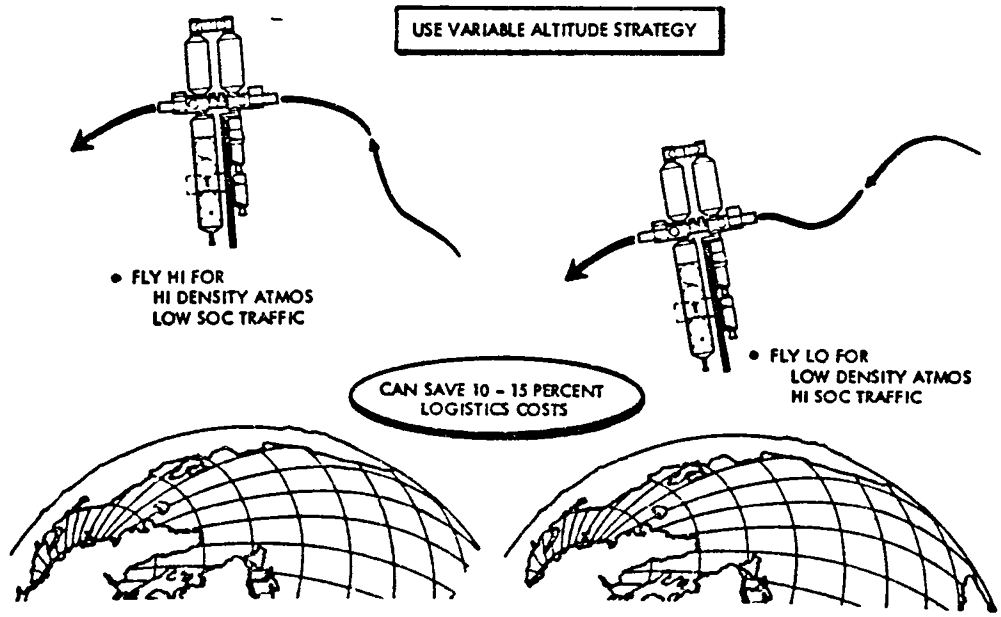

FIGURE 1.2 SOC ORBIT ALTITUDE STRATEGY

Direct Shuttle Delivery is the Recommended SOC Logistics Mode

Three basic SOC logistics modes were investigated. They are: (1) direct shuttle delivery in which all cargo is carried direct to the SOC in the orbiter bay, (2) tug assisted delivery in which all SOC cargo plus tug propellants are delivered by the Shuttle to 275 Km (150n.mi, maximum shuttle altitude with 65K lb. payload) where they are transferred to a SOC based tug for subsequent delivery to the SOC and (3) an OTV flydown mode where the basic SOC resupply cargo is still delivered direct to SOC by the shuttle, but OTV propellants and payloads are delivered to 275 Km where they are transferred to an OTV which was earlier flown down from the SOC. In this mode the OTV always returns to the SOC for servicing after a GEO mission and is flown down to the 275 Km refueling altitude at the beginning of the next mission.

The overall logistics performance of these three modes is nearly equal (less than 2% spread) over a wide range of traffic levels. The extra rendezvous ΔV's for the tug mission profile and the occasional extra shuttle flight to return the tug for ground refurbishment nearly negates the theoretical gains from not flying the heavy orbiter all the way to SOC altitudes. Similar negating factors occur with the OTV flydown mode including the extra ΔV required to stage GEO missions from 275 Km over that from higher altitudes.

Thus, because of the nearby equal performance of all modes and the major operational complexities introduced by the alternative modes, direct shuttle delivery is recommended for SOC.

1.2 THE SOC CONFIGURATION MODEL

The fully operational SOC configuration as illustrated in Figure 1.3 and the corresponding aerodynamic characteristics were used for the SOC orbit analysis. The SOC was considered to be normally flown in an Earth oriented attitude. This flight mode is with the -Z axis always pointing towards the Earth and the +X axis along the orbital velocity vector. However, the large solar arrays, that dominate the aerodynamic drag, are essentially inertially oriented in order to be able to track the sun for maximum effectiveness. The instantaneous frontal area exposed to the free molecular flow is thus not only dependent on the location of the SOC in its orbit, but is also dependent on the location of the sun with respect to the SOC orbit plane.

The greatest solar array area exposed to the flow would occur when the sun is in the orbit plane. The frequency in days of this geometric alignment is given by

$$FNSA = \frac{180}{1 + \dfrac{2.3825X10^{13}\cos i}{R_{SAT}^{3.5}}}$$

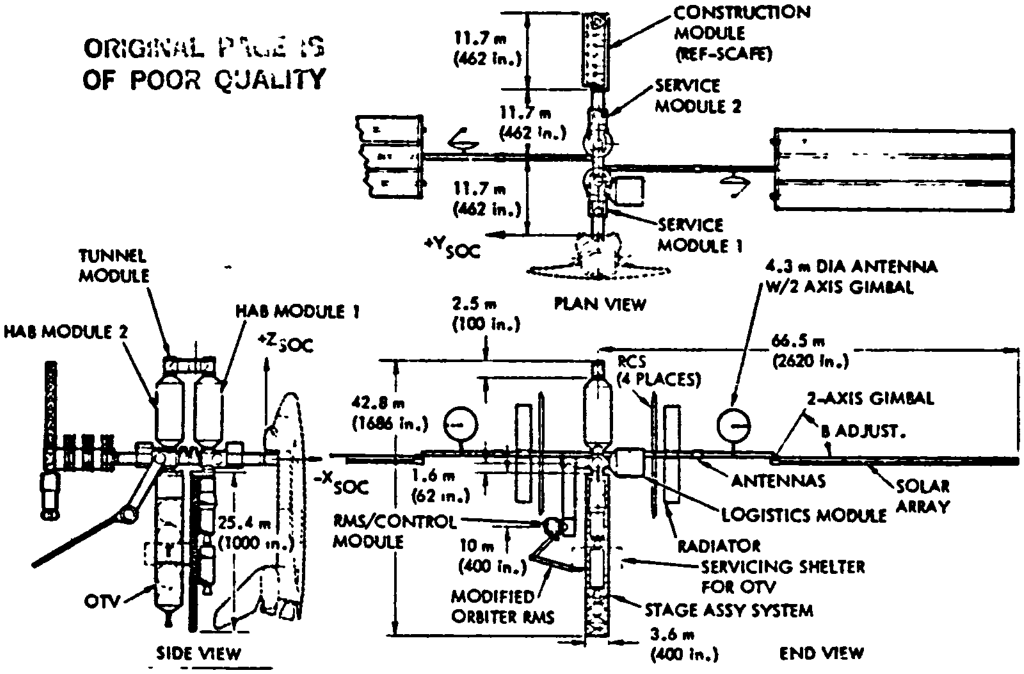

FIGURE 1.3 SOC REFERENCE CONFIGURATION

For a low altitude (180-400 km) orbit inclined at 28.5 degrees to the equator this event would occur approximately every 20-22 days.

The average ballistic coefficient for the SOC including the above factors was determined using the JSC provided data in Tables 1.1 and 1.2. Table 1.1 shows the aerodynamic force and moment coefficients for the complete SOC configuration as a function of angle of attack (α). This presumes that solar arrays remain fixed in the X-Y body axis plane. Thus, the effects of solar array rotation to maintain sun tracking are not included. These effects are presented in Table 1.2. The overall, long-term average drag coefficient for SOC was determined in two basic steps, the first to calculate the solar array sun tracking effects for a sun β-angle of 0 and then to incorporate the effects of β-angle variations (0 to 52 degrees for an orbit inclination of 28.5 degrees).

For β=0 the average "per orbit" drag coefficient can be expressed as:

$$C_{D_{avg}} = C_{D_{\alpha=0}} + \frac{2}{\pi} \left(\Delta C_{D_{\alpha=90}} - \Delta C_{D_{\alpha=0}}\right) \text{ solar array}$$

Where:

$C_{D_{\alpha=0}}$ is the total SOC drag at $\alpha=0$

$C_{D_{\alpha=0}}= 3.6266$ (from Table 1.1)

$\Delta C_{D_{\alpha=0}}$ is the drag of one side of the solar array at $\alpha=90^\circ$

$\Delta C_{D_{\alpha=90}} = 4.0888$ (from Table 1.2)

$\Delta C_{D_{\alpha=0}}$ is the drag of one side of the solar array at $\alpha=0^\circ$

$\Delta C_{D_{\alpha=0}} = 0.4312$ (from Table 1.2)

$\frac{2}{\pi}$ is the sine wave averaging factor.

As indicated above, the ΔC_D values in Table 1.2 are for half of the solar array and must be doubled to determine the total drag coefficient. Thus, the average drag coefficient for the total SOC configuration at $\beta=0$ is:

$$C_{D_{avg}} = 3.6266 + \frac{2}{\pi}(4.0B68-0.4312)x2 = 9.146 \quad (\beta = 0)$$

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TABLE 1.1 SOC AERODYNAMIC CHARACTERISTICS FOR REFERENCE CONFIGURATION

g) ROLL ANGLE = 0°

AREA = 2690 FT^2 L_{REF} = 58.52 FT ALTITUDE = 265 NMI

MOMENT REFERENCE CENTER AT X, Y, Z = 0, 0, 0

ALPHA	C_A	C_N	C_Y	C_D	C_m	C_n	C_ℓ
0 0	3 6266	0 0004	-0 0027	3 6266	-0 2928	0 1519	-0.0005
10 0	4 0537	0 9591	-0 0026	4.1585	-0 9153	0 1480	-0 0240
20.0	4 9247	2 1378	-0 0046	5 3588	-1 6157	0 1386	-0 0447
30.0	5 4216	3.5639	-0.0049	6.4771	-2 4657	0.1191	-0 0604
40.0	5 4574	5 0823	-0 0059	7 4473	-3 3861	0 0969	-0 0694
50.0	5.1140	6.6458	-0 0018	8 3780	-4.4730	0.0604	-0 0675
60.0	4.2581	7 9616	-0 0017	9 0239	-5 3411	0.0287	-0.0462
70.0	3.0[illegible]	8.9220	-0.0032	9 4214	-5.9535	0 0127	-0 0250
80 0	1 5860	9 4771	-0 0044	9 6083	-6.3169	0 0048	-0 0026
90.0	-0.0023	9 7807	-0 0036	9 7807	-6 5663	0.0002	0.0151
100 0	-1 5747	9.4319	-0 0012	9 5619	-6.3719	-0 0073	0 0009
110.0	-2 9943	8.8179	0 0013	9.3101	-6 0677	-0 0204	-0 0184
120 0	-4 2093	7 8718	0 0026	8.9217	-5 4228	-0 0419	-0 0409
130 0	-5 0706	6 5900	0 0006	8 3074	-4.5114	-0 0645	-0 0558
140 0	-5 4567	5 0734	0 0035	7 4411	-3 3760	-0 0934	-0 0600
150 0	-5 4099	3.5462	0 0049	6 4582	-2 2841	-0.1136	-0 0521
160 0	-4 8937	2 1094	0 0020	5 3200	-1 2155	-0 1296	-0 0405
170 0	-4 0348	0 9464	0 0026	4 1378	-0 3372	-0 1482	-0 0239
180 0	-3 6592	0 0011	0 0083	3 6592	0 3429	-0 1603	-0 0036
190.0	-4 0772	-0 9558	0 0043	4 1812	0 9099	-0 1474	0 0170
200 0	-4 8856	-2 1161	0 0049	5 3147	1 5607	-0 1474	0 0381
210 0	-5 4285	-3.5605	0 0011	6 4814	2 4070	-0 1248	0 0538
220 0	-5 5281	-5 1280	-0 0028	7 5309	3 3984	-0 0901	0 0602
230.0	-5 1591	-6 6843	-0 0055	8 4366	4 5034	-0.0664	0 0653
240 0	-4 2772	-7 9724	-0 0056	9 0429	5 3279	-0 0356	0 0469
250 0	-3 0271	-8 8930	-0 0017	9 3920	5 8867	-0 0185	0 0238
260 0	-1 5667	-9 4171	-0 0011	9 5463	6 2379	-0 0052	-0 0060
270.0	-0 0014	-9 6792	-0 0050	9 6792	6 4765	0 0020	-0 0261
280 0	1 5599	-9 3960	-0 0033	9 5240	6 3863	0 0034	-0 0035
290.0	2 9739	-8 7731	-0 0050	9 2616	6 0797	0 0127	0 0197
300 0	4 1849	-7 8346	-0 0029	8 8774	5 4 16	0 0262	0 0378
310 0	5.0581	-6 5863	-0 0033	8 2965	4 5964	0 0528	0 0528
320 0	5 4306	-5 0617	-0 0068	7 4136	3 4880	0 0863	0 0594
330 0	5 3861	-3 5400	-0 0054	6 4344	2 3693	0 1070	0 0526
340 0	4 8112	-2 0940	-0 0053	5 2372	1 3157	0 1253	0 0388
350 0	4 0138	-0 9497	-0 0039	4 1177	0 4094	0 1455	0 0219
360 0	3 6266	0 0004	-0 0027	3 6266	-0 2928	0 1519	-0 0005

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TABLE 1.2 SOC SOLAR PANEL AERODYNAMIC CHARACTERISTICS

$AREA_{REF}$ = 2690 FT^2 L_{REF} = 58.52 FT ALTITUDE = 265 N M.

MOMENT REFERENCE CENTER AT X, Y, Z = C, 0, 0

PANEL	α	ϕ	C_A	C_N	C_Y	C_D	C_m	C_n	C_ℓ
-Y SIDE	0°	0°	4312	0	0	4312	.0278	-1.1420	0
	30°		1 5884	1 1670		1 9592	- 6116	-4 2071	3 0909
	60°		1 5884	3 2159		3 5013	-1 8027	-4 2070	8 2789
	90°		0	4 0888		4.0888	-2 4860	0	10 8289
	0°	30°	.4312	0	0	4312	.0277	-1 1420	0
	30°		1 3758	.9149	- 3972	1 6869	- 4714	-3 4024	2.3986
	60°		1 3756	2 3986	-1 1913	3 0025	-1 3735	-2 9191	6 2791
	90°		0	3 1258	-1 5884	3 5012	-1 9006	9658	8 1809
	0°	60°	4312	0	0	4312	0278	-1.1420	0
	30°		8029	.3939	- 4014	9675	- 1900	-1 8822	1.0185
	60°		7943	9149	-1 1915	1 6870	- 5073	-1.3794	2.3497
	90°		0	1 1014	-1.4991	1 3490	- 6733	.9164	2 8751
+Y SIDE	0°	0°	4312	0	0	4312	0278	1 1420	0
	30°		1 5885	1 1671		1 9592	- 7280	4 2072	-3 0909
	60°		1 5885	3 1258		3 5013	-2 1142	4.2071	-8 2788
	90°		0	4 0887		4.0887	-2 8935	0	-10 8290
	0°	30°	4312	0	0	4312	0278	1 1420	0
	30°		1 3758	9149	- 3972	1 6869	- 5526	3 9249	-2.4477
	60°		1 3756	2 3985	-1 1913	3 0025	-1 6126	4 4865	-6 4261
	90°		0	3 1258	-1.5884	3.5013	-2 2122	1 1241	-8.3768
	0°	60°	4312	0	0	4312	0278	1 1420	0
	30°		8028	.3939	- 4014	9676	- 2293	2 4104	-1 0680
	60°		7944	9149	-1 1915	1 6869	- 5985	2 9470	-2 4966
	90°		0	1 1670	-1 5885	1 9592	- 8259	1 1242	-3 1889

The effects of sun β-angles can be determined using a similar sine wave averaging process, but with β-angle as the variable parameter. Since β-angle changes affect only the solar array drag we can initially ignore the non-array drag components. These will be added later. Thus, for the array only, the overall average drag coefficient including both the effects of orbital motion and β-angle variations may be expressed as:

$$\Delta C_{D_{avg}} = \Delta C_{D_{avg_{\beta=52°}}} + \frac{2}{\pi}\left(\Delta C_{D_{avg_{\beta=0}}} - \Delta C_{D_{avg_{\beta=52°}}}\right)$$

Where:

$\Delta C_{D_{avg_{\beta=52°}}}$ is the per orbit average drag of one side of the solar array at β=52° (maximum β)

$$\Delta C_{D_{avg_{\beta=52°}}} = \Delta C_{D_{\alpha=0}} + \left(\Delta C_{D_{avg_{\beta=0}}} - \Delta C_{D_{\alpha=0}}\right) \cos 52°$$

$\Delta C_{D_{\alpha=0}}$ is the drag of one side of the solar array at α=β=0

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$\Delta C_{D_{\alpha=0}} = 0.4312$ (from Table 1.2)

$\Delta C_{D_{avg_{\beta=0}}}$ is the per orbit average drag of one side of the solar array at $\beta=0$

$$\Delta C_{D_{avg_{\beta=0}}} = \Delta C_{D_{\alpha=0}} + \frac{2}{\pi}(\Delta C_{D_{\alpha=90}} - \Delta C_{D_{\alpha=0}})$$

$$\Delta C_{D_{avg_{\beta=0}}} = 0.4312 + \frac{2}{\pi}(4.0888 - 0.4312)$$

$$\Delta C_{D_{avg_{\beta=0}}} = 2.7597$$

Thus,

$$\Delta C_{D_{avg_{\beta=52^\circ}}} = 0.4312 + \frac{2}{\pi}(2.7597 - 0.4312) = 1.8648$$

The overall average drag coefficient of the solar array then becomes

$$\Delta C_{D_{avg}} = [1.8648 + \frac{2}{\pi}(2.7597 - 1.8648)] \times 2 \text{ solar array sides}$$

$$\Delta C_{D_{avg}} = 4.869$$

Adding the non-array drag components from Table 1.1 the total SOC drag coefficient including the averaged effects of orbital motion and β-angles becomes

$$C_{D_{SOC_{avg}}} = C_{D_{SOC_{\alpha=0}}} + \Delta C_{D_{avg}}$$

$$C_{D_{SOC_{avg}}} = 3.6266 + 4.869 = 8.4956$$

The above drag coefficients from Tables 1.1 and 1.2 are based on a reference area of 2690 sq ft . The average ballistic coefficient for SOC becomes:

$$BC_{SOC} = \frac{W}{C_D A} = \frac{245000}{(8.4956)(2690)} = 10.72 \text{ psf or } 52.33 \text{ Kg/m}_2$$

This ballistic coefficient value is for the basic SOC system orthagonally trimmed to a local horizontal attitude. Thus, changes in trim attitude, the presence of OTV's and/or construction projects and the presence of the orbiter during resupply operations can all affect the SOC operational ballistic coefficient.

These effects were briefly examined and found to be small and/or tending to increase the ballistic coefficient. First, trim angle effects were considered. Analysis of angular momentum buildups in Section 3.0 of this report indicated the desirability of operating SOC in trim attitudes which use gravity-gradient torques to balance secular aerodynamic torques. Because of configuration symmetry these trim attitudes are mostly pitch angle adjustments within the orbit plane. Drag effects of these trim angles can be approximated using the data in Tables 1.1 and 1.2. Subtracting the solar array drag values in Table 1.2 from the total configuration drag in Table 1.1 for angles of attack of 30 and 90 degrees indicates the drag coefficient of the central modular cluster actually decreases slightly with increased angle of attack. This is apparently due to the reduced frontal area of the vertically oriented surfaces (habitability modules, radiators, and stage assembly module) which dominates the slight increase in frontal area of the horizontal configuration elements (service modules). These delta drag effects are small, less than 5 percent for trim angles up to 30 degrees and would tend to increase the SOC ballistic coefficient.

Future OTV configurations and space construction projects which will be supported by the SOC cannot be precisely defined at this time. However, certain generalized characteristics can be examined. For example, cryogenically propelled OTV's with dimensional limits permitting transport in the orbiter cargo bay (15 feet diameter, etc.) produce ballistic coefficients of about 50 Kg/m^2 (10 psf) when vertically mounted on the SOC (their full length exposed to the free stream flow). When filled with propellants their ballistic coefficient increases to the 350 Kg/m^2 (70 to 80) psf range. OTV missions are expected to last from a few days to a couple of weeks. Thus, most of the time the OTV will be stored empty on the SOC. Part of the remaining time it will be in increasing states of propellant fill conditions. The rest of the time, of course, it will be on active mission status away from the SOC. Thus, the main effect of OTV's on SOC drag will be to increase the overall ballistic coefficient during the refueling periods; the rest of the time it has virtually no influence.

Previous studies of space construction projects give some indication of their potential influence on the SOC ballistic coefficient. For example, just the basic structure of a space fabricated tri-beam platform was determined to have a ballistic coefficient of approximately 35 K_g/m^2 (7 psf). This could increase to 100 Kg/m^2 (20 psf) or more with the installation of relatively dense subsystems modules. Thus, the likely range of ballistic coefficients introduced by space construction projects also appears to be near that of the basic SOC configuration and can probably be ignored for this preliminary analysis of orbit altitude.

The orbiter, while docked to the SOC, is oriented in its maximum drag configuration. Even with this orientation its ballistic coefficient is of the order of 150 Kg/m^2 (30 psf), which is somewhat higher than the basic SOC. Although the combined SOC-orbiter ballistic coefficient would fall in the 100 Kg/m^2 (20 psf) range, the residence time of the orbiter is small so its long term average influence will be small.

Thus, all four of these configuration variables have relatively small effects on the long term SOC ballistic coefficient and what little effects there are tend to increase it. Therefore, a slightly conservative value of 50 Kg/m^2 (10 psf) was used for the SOC ballistic coefficient in this study.

1.3 STS PAYLOAD PERFORMANCE

Shuttle payload performance is a major driver in the optimum logistics altitude for SOC operations. In particular, the rate of payload drop-off with altitude (the slope $[\Delta P/L]/\Delta h$) is the critical parameter. In general, the steeper the slope of this drop-off the lower the optimum altitude will be. This is logical since the optimum altitude is a balance between the need for more Shuttle flights at high altitudes due to reduced payload capability and the need for more orbit makeup propellants at low altitudes.

STS performance characteristics for use in the study were obtained from JSC and are presented in Figure 1.4. Both the standard orbiter and a Titan liquid boost module (LBM) thrust augmentation capability are shown. These data were generated using a three-dimensional, three degree-of-freedom trajectory program including all of the necessary trajectory shaping criteria and constraints for KSC launches. The JSC package containing the detailed conditions and guidelines used for these data are contained in Appendix C to provide a complete source for future reference.

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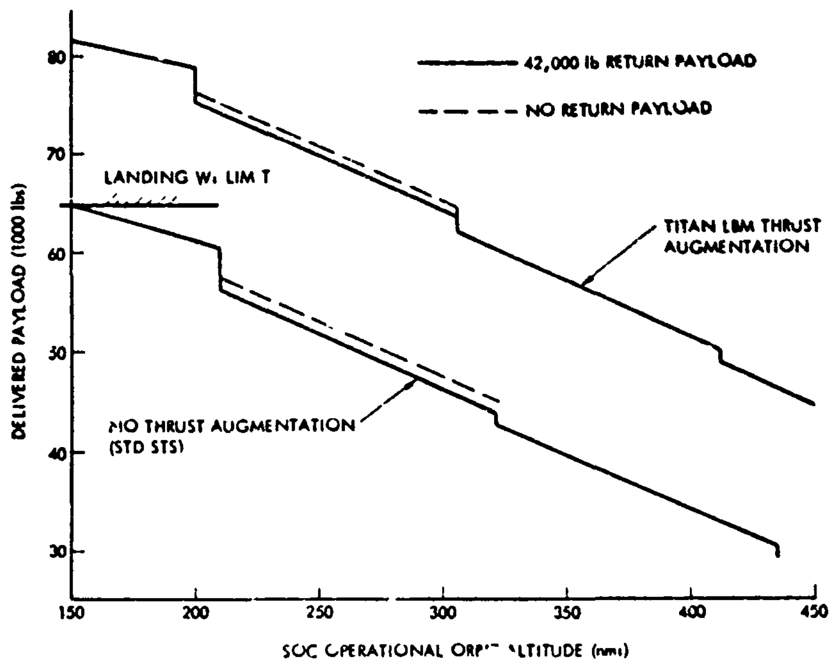

FIGURE 1.4 STS PAYLOAD PERFORMANCE

As derived from the above curves the payload drop-off rates with altitude are minus 63 Kg/Km (75 lb/n.mi) and minus 50 Kg/Km (59 lb/n.mi) for the standard and thrust augmented STS respectively. In the subsequent orbit altitude analysis the landing weight limit of 29500 Kg (65,000 lb) was not imposed on the performance capability of the thrust augmented STS. However, if this landing weight limit cannot be increased for the augmented configuration the optimum logistics altitude would correspond to the highest altitude attainable with a 29,500 Kg payload (somewhere near 550 Km or 300 n.mi). This is because at any altitude below this value orbit makeup propellant requirements will be greater without any offsetting gain in Shuttle payload capability.

1.4 ORBIT DECAY CONSIDERATIONS

The SOC complex will be situated in a low earth orbit (275-550 Km) within the Shuttle performance envelope. Although the atmosphere at these altitudes is tenuous a spacecraft as large and complex as the SOC would still experience considerable atmospheric drag. Thus, orbit lifetime is one of the most important factors in the determination of SOC orbit altitude. As a contingency margin in the event that orbit makeup capability is temporarily lost a suitable orbit decay life will assure against catastrophic reentry and the corresponding worldwide debris hazard.

The variables that affect the calculations of orbital lifetime, orbit altitude decay rate, and orbit altitude maintenance propellant requirements fall in three general categories.

The first group of the parameters is spacecraft dependent. These are the mass or weight of the vehicle and the area of the spacecraft that is exposed to the flow of the atmosphere around it. The detailed characteristics of the exposed area such as temperature and geometry can also affect the drag coefficient of the body. However, for use in this study the average ballistic coefficient for the full-up developed SOC, as derived in Section 1.1 is 50 Kg/m^2 (10 psf).

The second group of parameters affecting decay life is orbit dependent. The speed or velocity of the spacecraft and the atmospheric density both have great influence on satellite lifetime in orbit.

Finally the third group, if it can be called such, characterizes the propulsion subsystem that will be employed to maintain the spacecraft in orbit. The efficiency of the propulsion subsystem is described by its specific impulse.

Of all the variables mentioned here that affect the calculations of orbit lifetime, altitude decay rate, and orbit altitude maintenance, the atmospheric density is most significant. This significance is reached not from its effect in the calculations, but rather from the difficulty to estimate the actual density at any given time. This difficulty arises from having to predict solar activity as a function of time.

It is for this reason that several atmospheric profiles were considered in the subsequent analyses.

In general, the solar activity cycle from peak-to-peak is approximately eleven years. The present peak (1979-1980) was extremely high, resulting in extreme atmospheric densities. The next predicted solar activity peak is for 1990-1991. Atmospheric density predicted for this 1990 period is shown in Figure 1.5. Both nominal and $+2\sigma$ atmospheres are shown. This data is for long term prediction based on NASA TM 53865.

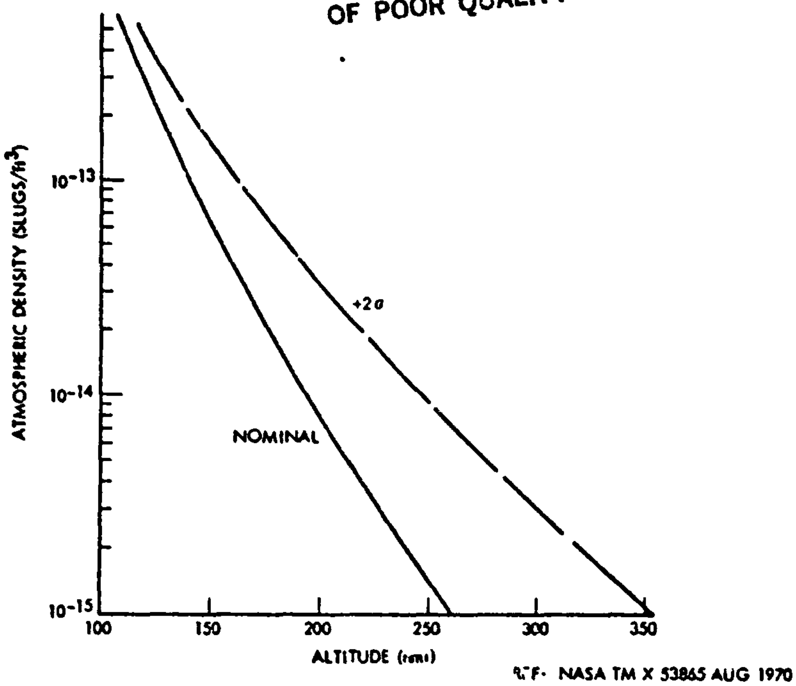

FIGURE 1.5 LONG TERM PREDICTION ATMOSPHERIC DENSITY, MAXIMUM SOLAR ACTIVITY - 1990

The two solar activity related parameters that affect the earth's atmosphere are the geomagnetic index and the solar flux. The solar flux level is measured or estimated at the 10.7 cm wavelength.

Actual measurement data for 1979 through mid-1980 are shown in Figure 1.6. The predicted statistical function for the next three and a half years is also illustrated in the figure. The corresponding atmospheric densities (nominal, $+2\sigma$, and $+3\sigma$) are shown in Figure 1.7. The nominal density curve for 1980 (Figure 1.7) closely approximates the $+2\sigma$ density predicted for 1990 (Figure 1.5). This is because the recent actual measured density has been extremely high, thereby biasing the near term statistical projections.

Finally, a Shuttle design atmosphere based on specific flux (230×10^{-22} watts/M^2/cycle/second) and geomagnetic index (20.3) values was estimated. This atmospheric density is shown in Figure 1.8 and it closely approximates the $+3\sigma$ values shown in Figure 1.7 for the latter part of 1980. This is the design "high density" atmosphere used in this study for establishing SOC requirements.

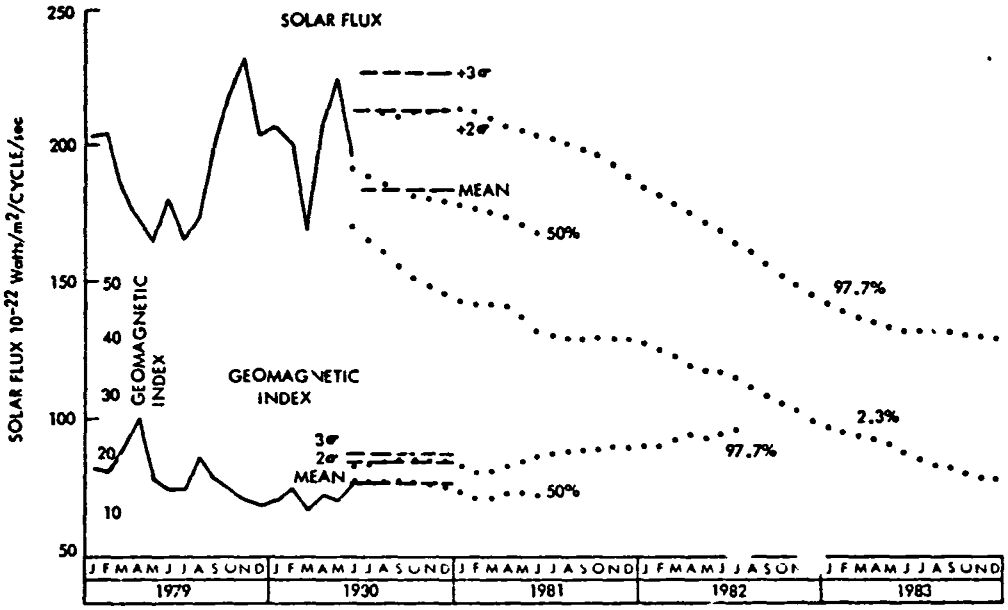

FIGURE 1.6 SOLAR ACTIVITY PREDICTION

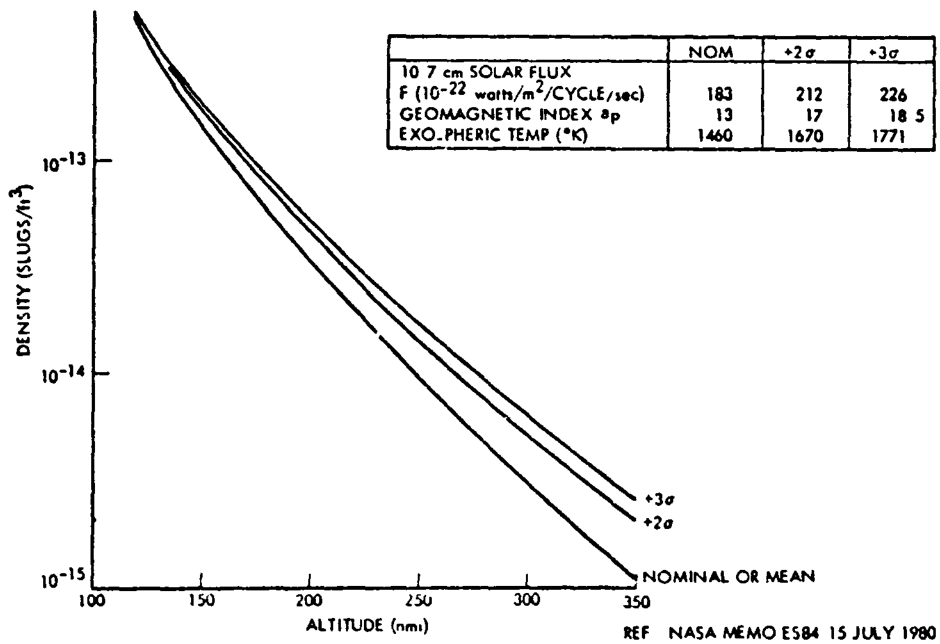

FIGURE 1.7 ATMOSPHERIC DENSITY, MAXIMUM SOLAR ACTIVITY, SECOND HALF OF 1980 (SHORT TERM PREDICTION)

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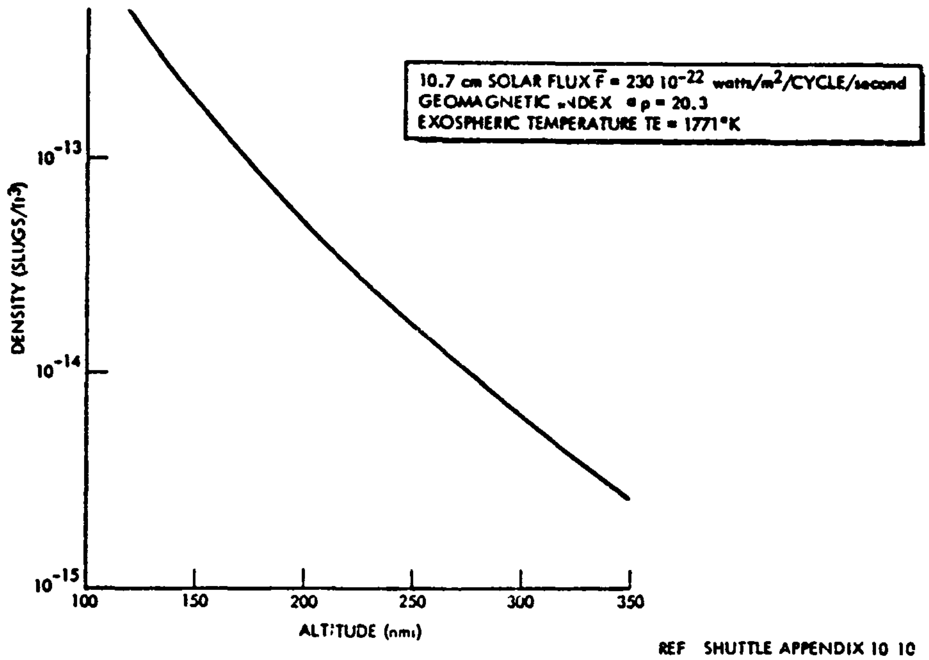

FIGURE 1.8 DESIGN ATMOSPHERIC DENSITY

To maintain a circular orbit in the presence of atmospheric drag the orbit makeup propellant is given by the relationship

$$WPR = \frac{t_M}{W/C_D A} \quad \frac{\rho\mu/2r}{I_{sp}} \quad W$$

where t_M is the mission lifetime or specifically the resupply interval and r the radius of the spacecraft orbit. An interesting observation that can be made is that the propellant requirement is independent of the spacecraft weight for a fixed average frontal area used in the definition of the ballistic coefficient.

$$WPR = t_M \; C_D A \;\; \rho\mu/2rIsp$$

For SOC where the average ballistic coefficient was taken to be 50 Kg/m^2 (10 psf) the above relationship reduces to

$$WPR = .993 \times 10^8 t_M \rho/rIsp$$

where t_M is in years
r is in nautical miles
ρ density in slugs/ft^3

The atmospheric parameter r/ρ is a function of the statistical analysis and prediction of the solar activity for a particular time period. For the various atmospheric models previously discussed the yearly orbit maintenance propellant requirements are illustrated in Figure 1.9. The type of atmosphere predicted can result in nearly an order of magnitude variation in the propellant required for orbit maintenance. The effect of specific impulse on the propellant weight to be carried is illustrated in Figure 1.10. The higher the specific impulse, i.e., more efficient propulsion system, the lower the orbit maintenance requirements.

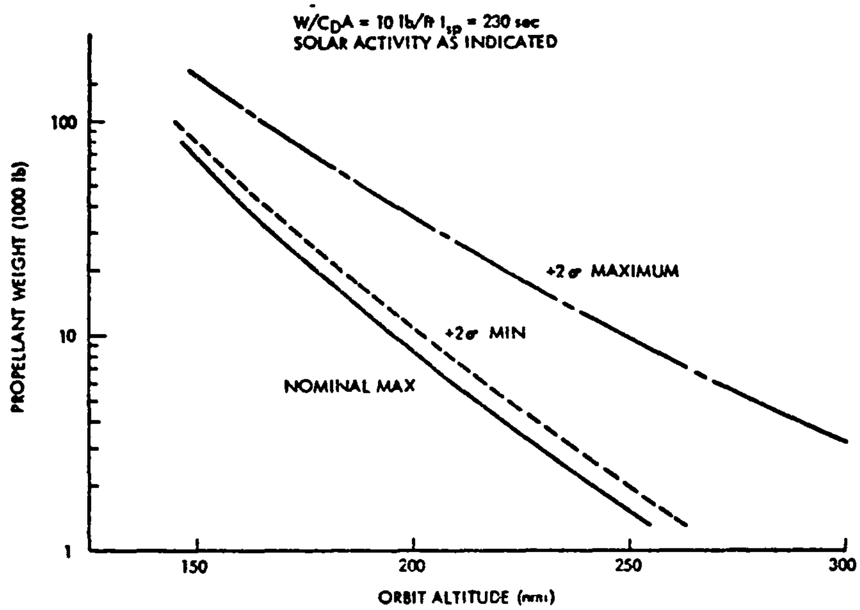

FIGURE 1.9 YEARLY PROPELLANT REQUIREMENTS TO MAINTAIN SOC ALTITUDE

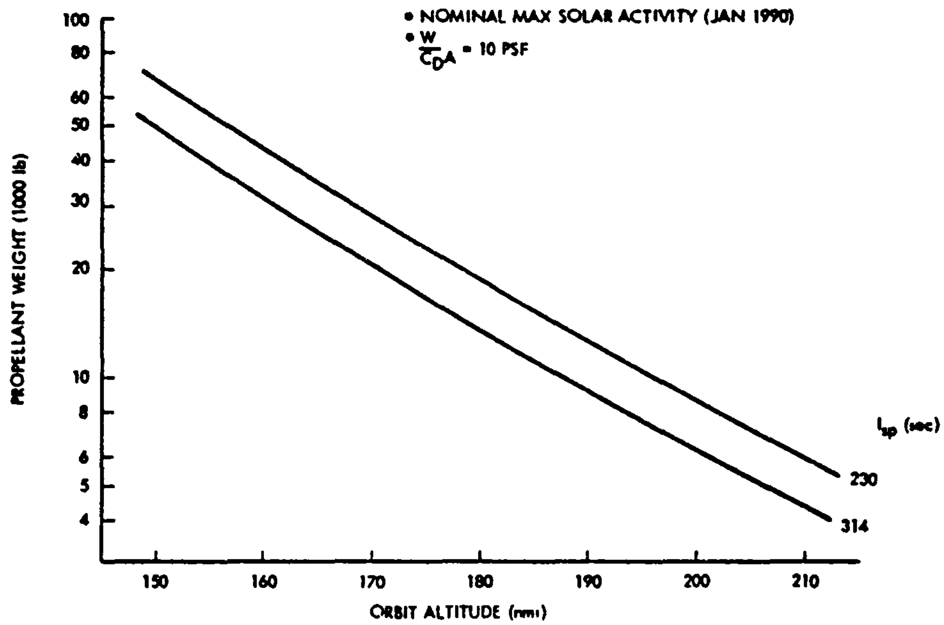

FIGURE 1.10 I_{sp} EFFECTS ON YEARLY PROPELLANT REQUIREMENTS

Although orbit makeup propellant is important, system safety is the primary driver for orbit altitude selection. Hence, the time to decay to a critical altitude becomes of prime concern. The time it would take the entire SOC complex to decay to a 185 Km (100 n.mi) altitude will be used to define a decay safe orbit. This time to decay - without any orbit maintenance is illustrated in Figure 1.11. Again, it is seen to be an extremely strong function of the atmospheric model and hence our ability to predict the future.

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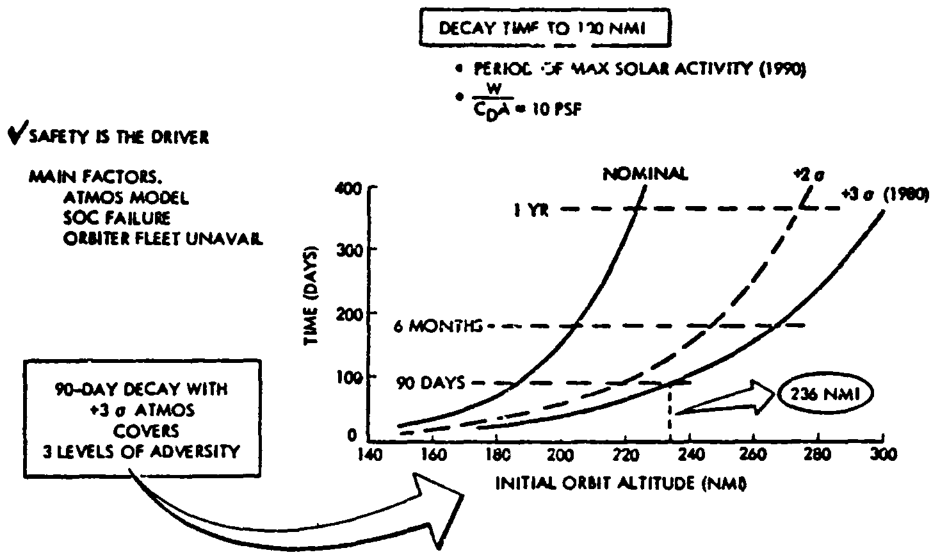

FIGURE 1.11 ORBIT DECAY CONSIDERATIONS

For example, with the nominal atmosphere predicted for 1990 a 377 Km (204 n.mi) altitude would be sufficient so that it would take six months for an unmaintained SOC configuration to decay to a 185 Km (100 n.mi) altitude. However, a +2σ atmosphere for the same time period would decrease the decay time to just 60 days or require the orbit to be raised to 451 Km (247 n.mi) to maintain the six month decay time. However, it is interesting to note that for the altitudes of constant orbit lifetimes (90,180 or 360 days) the corresponding orbit maintenance propellant is only a weak function of the atmospheric model. This relationship is illustrated in Figures 1.12 and 1.13. Thus, it is possible to define the orbit maintenance propellant requirement based on a safe orbit decay time.

An example is illustrated in the "box" on Figure 1.12. A variable altitude strategy is assumed for this example. This means propellant for maneuvering to the desired altitudes must be determined in addition to that required for basic orbit makeup. As a worst case it was assumed that the atmospheric density shifted from a nominal condition to a +3σ value during the time of maximum solar activity. For this situation an orbit altitude increase of 93 Km (50 n.mi, from 185 n.mi to 235 n.mi) would be required to maintain the 90 day decay life criteria. This would require 2500 Kg (5500 lb) of propellant at the consumption rate of 28 Kg per Km (114 lb per n.mi) for this maneuver.

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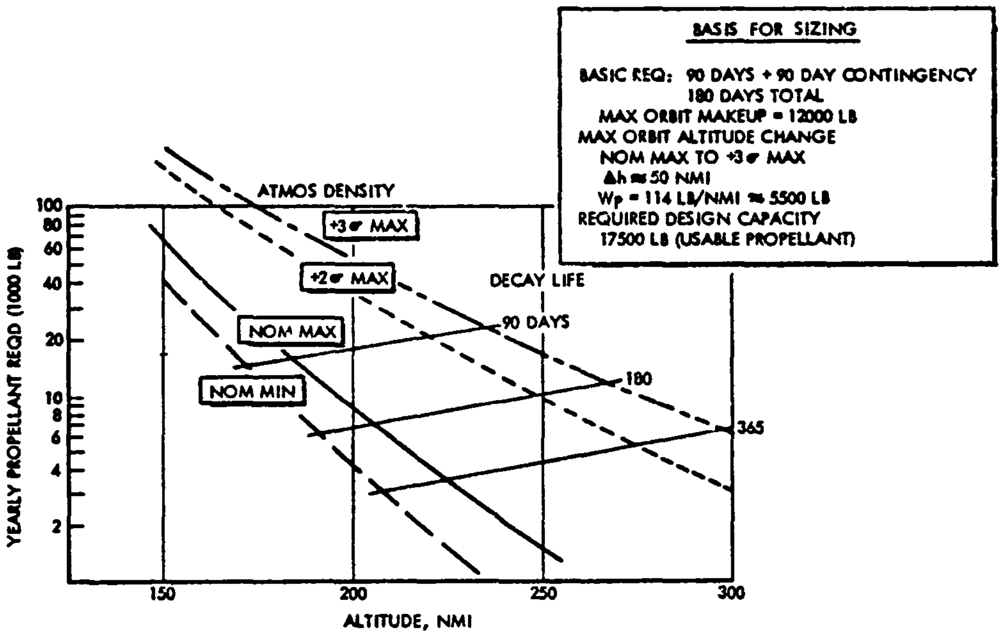

FIGURE 1.12 SOC PROPELLANT SIZING

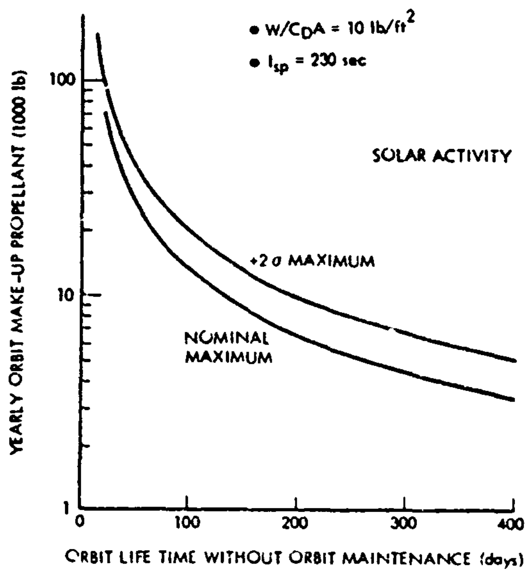

FIGURE 1.13 ORBIT MAINTENANCE VS LIFETIME

Since the atmospheric density tends not to change rapidly (aside from small daily perturbations, major density changes tend to occur over a period of months rather than days) it was conservatively assumed for this example to maintain the +3σ condition for the entire 180 day propellant sizing requirement. Therefore, the total propellant capacity requirement for the SOC would be 7950 Kg (17,500) lb, 5450 Kg for the worst case orbit makeup plus the 2500 Kg for the worst case altitude change maneuver to maintain the 90 day decay criteria. Actually, the orbit change maneuver could have boen a series of maneuvers so long as the total altitude change did not exceed 93 Km. Any other set of conditions than the worst case assumptions used here would require less propellant.

It is highly unlikely that the timing of atmospheric density rises would occur such that both the worst case orbit makeup propellant and the worst case altitude change maneuver would be required during the SOC resupply interval. It is much more likely that at least part of the resupply interval would be spent at some lesser density condition, and hence use less orbit makeup propellant; or that the SOC had already been maneuvered or "partly maneuvered" to the higher orbit altitude prior to a resupply mission, and thereby not require the worst case maneuver propellant at the same time it needed the maximum makeup propellant. Thus, this example propellant sizing requirement is conservative.

1.5 LAUNCH OPPORTUNITY INTERVALS

The execution and planning of materials/equipment/propellant delivery and resupply missions to the Space Operations Center is more easily facilitated if a predictability repetitive orbit path is maintained. For example, an orbit that repeats its ground trace exactly every day will present an optimum resupply launch opportunity (in plane, in phase) on a daily basis. The relationship involves both orbit altitude and inclination. This is illustrated for an orbit inclination of 28.5 degrees in Figure 1.14. In the altitude region of greatest interest (275-350 Km) orbits with a trace repetition every two, three, or five days are feasible. At these low altitudes atmospheric drag will reduce the orbit altitude, thereby modifying this repetitive relationship. To maintain the trace repetition within some specified limits certain frequencies of orbit makeup impulses must be maintained. The frequencies to maintain a ground trace within a ±10 km crosstrack (out of plane) deviation are shown in Figure 1.15 as a function of orbit altitude and spacecraft ballistic coefficient. For the SOC ballistic coefficient of 50 Kg/m^2 (10 psf), the orbit maintenance corrective maneuvers would have to be made no more frequently than once every two days even for orbits as low as 275 Km (150 n mi). Since other spacecraft propulsive operations (momentum dump) must be performed more frequently the orbit ground trace or altitude maintenance maneuvers could be accomplished at the same time.

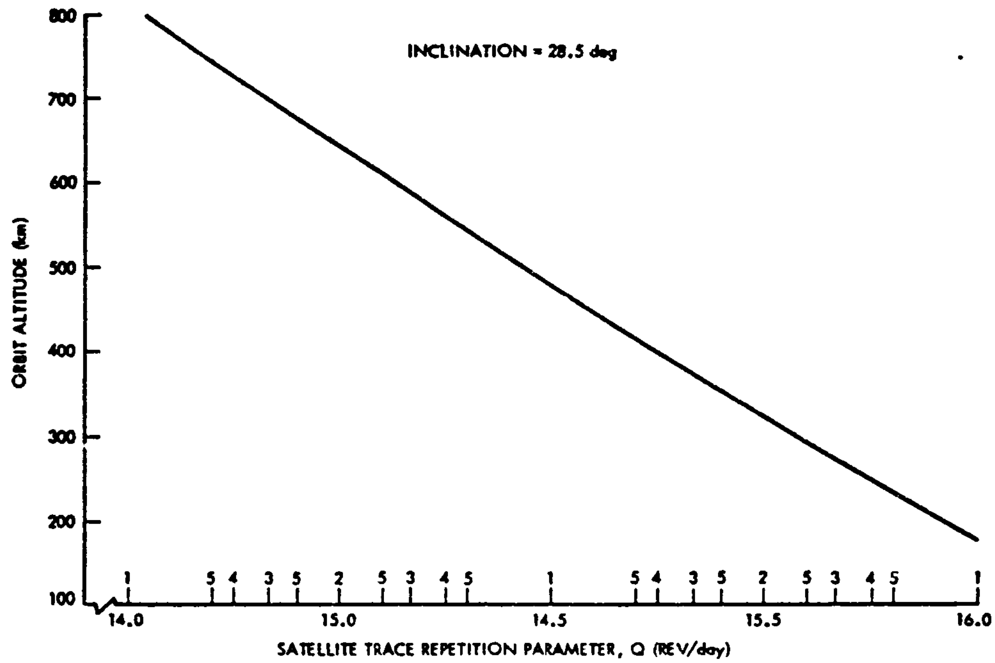

FIGURE 1.14 SATELLITE GROUND TRACE REPETITION

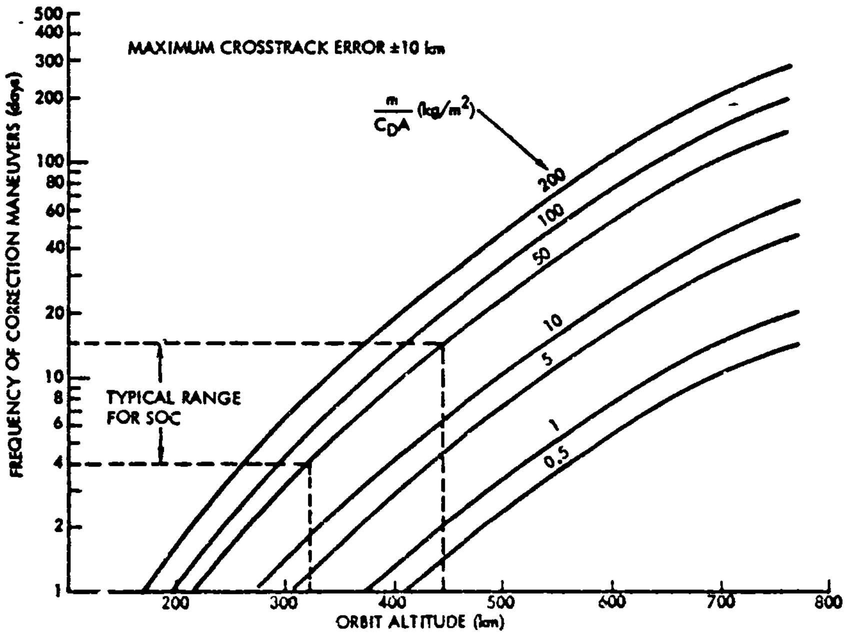

FIGURE 1.15 FREQUENCY OF CORRECTION MANEUVERS

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1.6 SOC LOGISTICS MODES EFFECTS

Three different SOC logistics delivery modes with the standard STS were investigated along with one mode for the thrust augmented STS to determine their comparative logistics efficiencies and their potential effects on SOC orbit altitude. The three basic modes which are pictured symbolically in Figure 1.16 are: (1) direct Shuttle delivery, (2) tug assisted delivery, and (3) OTV fly down delivery.

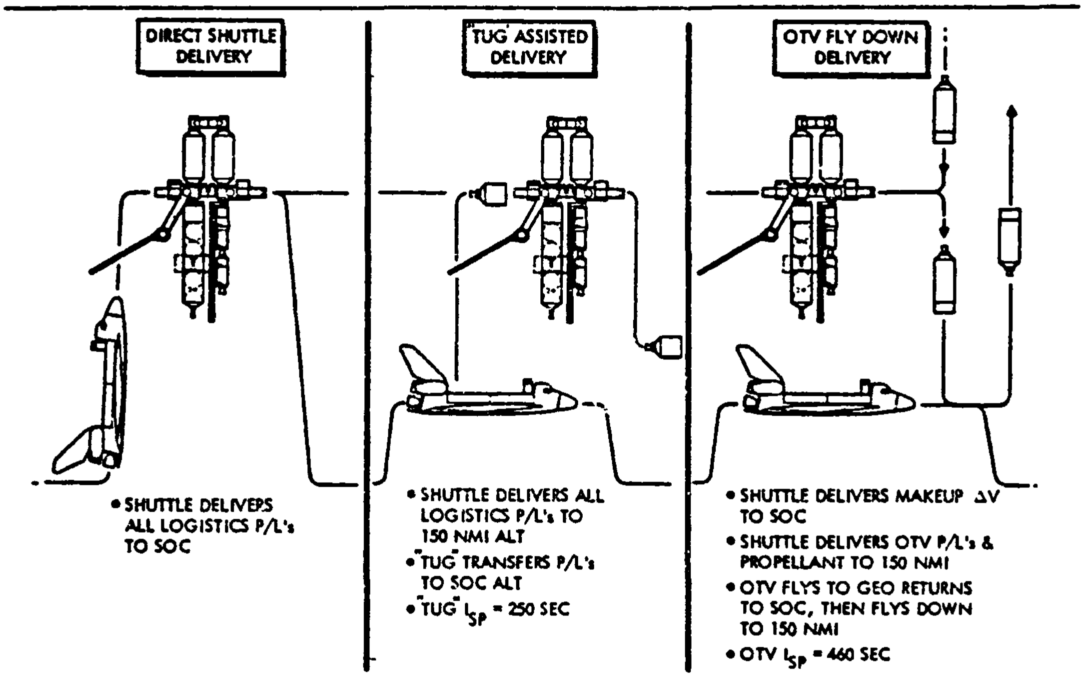


FIGURE 1.16 SOC LOGISTICS MODE OPTIONS

Mode Descriptions

With the direct delivery mode the Shuttle carries all SOC supplies direct to the SOC and exchanges them for any return cargo that might be required. The Shuttle is launched approximately coplaner with the orbiting SOC and performs ascent phasing and rendezvous maneuvers leading to docking with the SOC. After the delivered materials are off loaded and any return cargo is installed in the orbiter bay the Shuttle deorbits and returns to KSC.

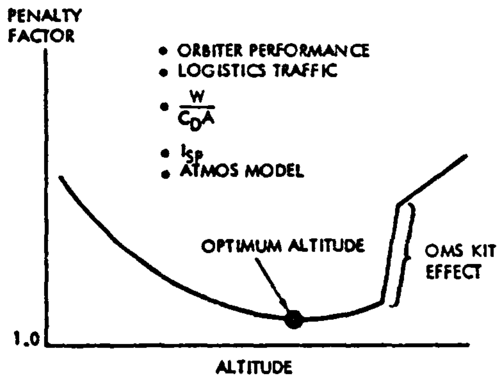
With the tug assisted delivery mode the Shuttle delivers all of the SOC cargo to 275 Km (150 n.mi) altitude. This is the maximum altitude to which a 29500 Kg (65K lb) payload can be delivered by the standard Shuttle (Figure 1.4). At the 275 Km orbit the "up" cargo is transferred to a SOC based tug for subsequent delivery to the SOC. Any "down" cargo required from SOC operations would be carried to the transfer point by the tug and exchanged for the "up" cargo. This would include normal SOC crew rotations, thus necessitating a "man rated" tug. The Shuttle then returns to KSC.

With the OTV flydown mode the normal SOC resupply cargo (crew expendables, operating fluids, etc.) is carried directly to the SOC with the Shuttle as in the direct delivery mode. However, OTV propellants (a major logistics component) and OTV payloads are delivered to the 275 Km altitude as in the tug assist mode. The OTV flys down from the SOC for refueling and payload installation. The OTV then performs its GEO mission and returns to the SOC for turnaround servicing. In the meantime after refueling the OTV, the orbiter returns to KSC. Thus, in this mode part of the SOC cargo goes direct to the SOC in the orbiter and the rest is delivered to the OTV at 275 Km altitude.

In addition to these three basic logistics modes with the standard STS the effects of the thrust augmented STS with the direct delivery mode were also investigated. Variations in SOC logistics traffic levels were analyzed for all cases.

Logistics Penalty Factor

A parameter called the logistics penalty factor was used as means of determining optimum SOC altitudes for comparing logistics modes. Its mathematical definition and characteristic plotted shape are shown in Figure 1.17. Essentially, the logistics penalty factor is a dimensionless quantity which normalizes the main factors which can affect the number of logistics flights required to deliver a given amount of cargo and allows easier comparison of modes and other variables. Since the specific manifest details of the SOC related cargo cannot be known this far in advance of actual SOC operations, the logistics penalty factor becomes a convenient parameter for compaiing delivery modes and determining optimum orbit altitudes.



PENALTY FACTOR DEFINITION

NORMALIZED TO NUMBER OF SHUTTLE FLTS REQUIRED TO DELIVER ASSUMED TRAFFIC TO 150 N MI

INCREMENT ABOVE 1.0 ACCOUNTS FOR SOC MAKEUP ΔV & SHUTTLE P/L DROP-OFF WITH ALTITUDE

$$\text{PENALTY FACTOR} = \frac{\left(1 + \dfrac{W_{MAKEUP}}{W_{TRAFFIC}}\right)}{\left(\dfrac{W_{STS_{ALT}}}{W_{STS_{150}}}\right)}$$

FIGURE 1.17 LOGISTICS PENALTY FACTOR DEFINITION

For these analyses the logistics penalty factor is normalized to the number of Shuttle flights required to deliver basic SOC cargo (excluding orbit makeup propellant) to 275 Km (150 n.mi) altitude. This is the highest altitude attainable by the standard Shuttle w tn a full payload (Figure 1.4). Thus, the denominator term in Figure 1.17,

$$\left(\frac{W_{STS}\ ALT}{W_{STS_{150}}} \right)$$

causes the penalty factor to increase as Shuttle payload drops off with altitude. If orbit makeup propellant were zero (and did not change with altitude) the penalty factor would simply increase with altitude from a value of 1.0 at 275 Km (150 n.mi) altitude.

However, orbit makeup propellant increases at the lower altitudes. This effect is contained in the numerator term of the penalty factor equation, which is the ratio of total logistics traffic (including makeup propellant) to just the basic traffic. Specifically, then, the logistics penalty factor is the ratio of the number of Shuttle flights required to deliver the total SOC cargo to any particular altitude to the number of Shuttle flights required to deliver just the basic cargo to 275 Km (150 n.mi). Thus, the increment above 1.0 accounts for both orbit makeup propellants and Shuttle payload dropoff with altitude and the altitude where the penalty factor is smallest in the optimum logistics altitude.

The factors which can affect the optimum logistics altitude are listed on the figure and include those things that influence the amount of orbit makeup propellant required (drag, Isp, etc.) and the delivery system performance. Inspection of the penalty factor equation will show that as the basic SOC logistics traffic is reduced the orbit makeup propellant factor will begin to dominate the optimum altitude selection. On the other hand at high traffic levels the Shuttle payload performanco dominates and in the extreme would optimise at the 275 Km (150 n.mi) maximum payload point.

Direct Delivery Mode

Figures 1.18 through 1.21 present the penalty factor plots for the direct delivery mode with the standard STS. SOC logistics traffic levels ranging from one-fourth to two SOC masses per year are presented. One SOC mass represents approximately 110,000 Kg (245,000 lb). This range of traffic levels reflects the logistics intensive nature of an operations orientated space base. Space construction and OTV operations both require heavy logistics support compared to experiments and technology development types of space activities. The data in these curves are based on a SOC ballistic coefficient (W/C_DA) of 50 Kg/m^2 (10 psf), an orbit makeup propulsion Isp of 230 sec and are for a nominal maximum atmospheric density condition (1990 nominal). As indicated in the discussion above, the optimum logistics altitude decreases for the high traffic levels because Shuttle performance dominates over the drag makeup effects.

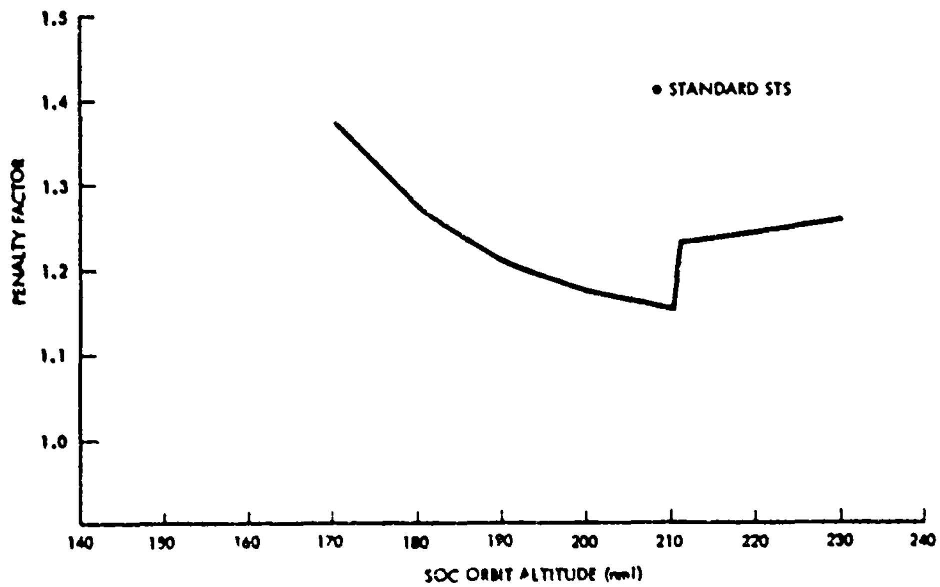


FIGURE 1.18 DIRECT DELIVERY MODE - ONE FOURTH SOC MASS PER YEAR

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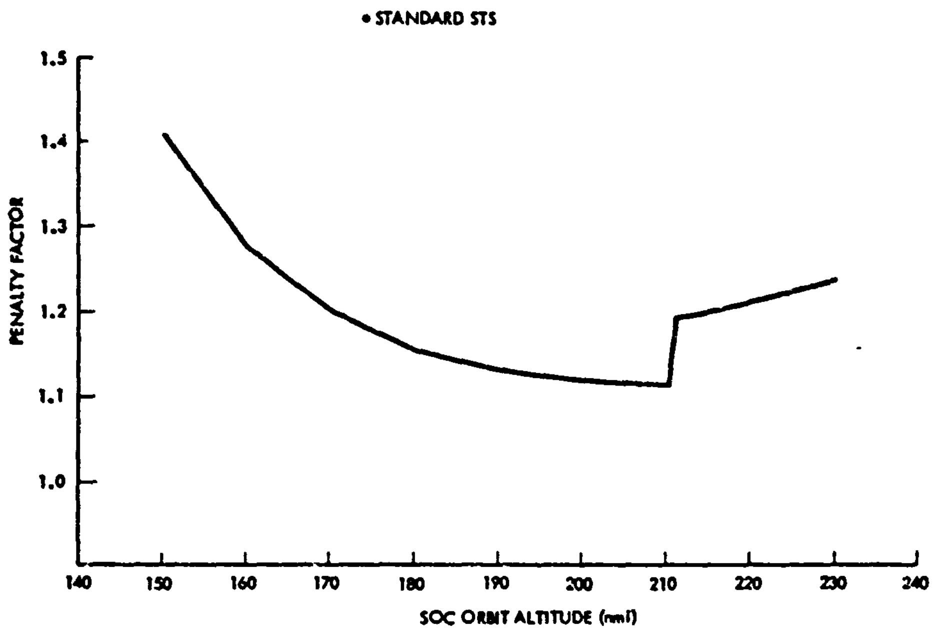

FIGURE 1.19 DIRECT DELIVERY MODE - ONE HALF SOC MASS PER YEAR

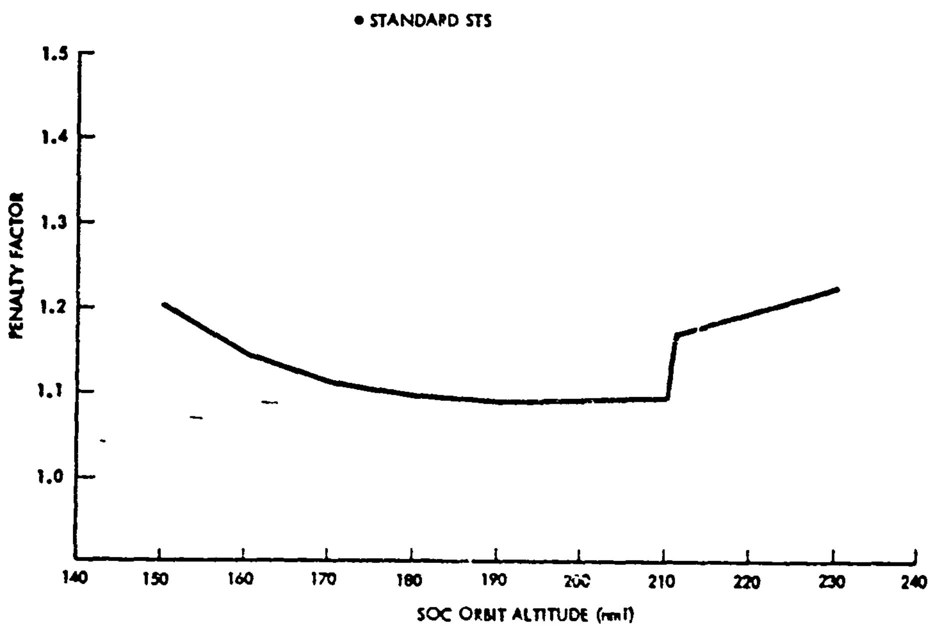

FIGURE 1.20 DIRECT DELIVERY MODE - ONE SOC MASS PER YEAR

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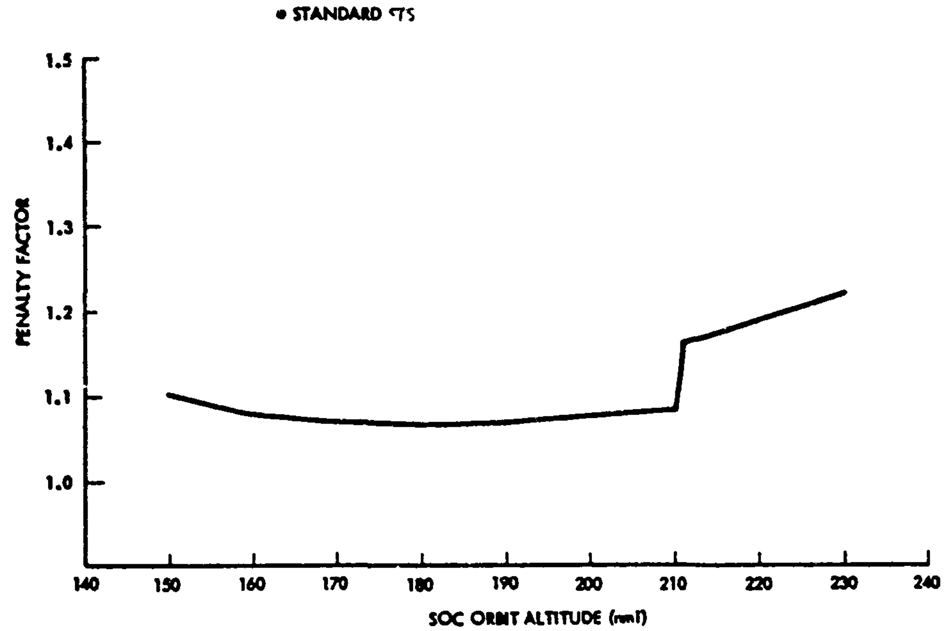

FIGURE 1.21 DIRECT DELIVERY MODE - TWO SOC MASSES PER YEAR

Figure 1.22 shows the effects of both atmospheric density variations and logistics traffic levels on the optimum SOC orbit altitude. As mentioned before, high traffic levels decrease the optimum altitude. Also, as would be expected, unusually high atmospheric densities tend to increase the optimum altitude. It is interesting to note that even for the extreme density case (+3σ max) the optimum altitude does not move above the 390 Km (210 n.mi) "barrier" introduced by the inert weight effects of adding an OMS kit. The basic penalty factor curves are sufficiently shallow in the vicinity of the optimum altitude that the 1360 Kg (3000 lb) inert weight increment for adding an OMS kit completely overpowers the gradual changes in makeup propellant and payload dropoff slope for the orbiter. Thus, with extreme densities the optimum logistics altitude for SOC would be constant at 390 Km.

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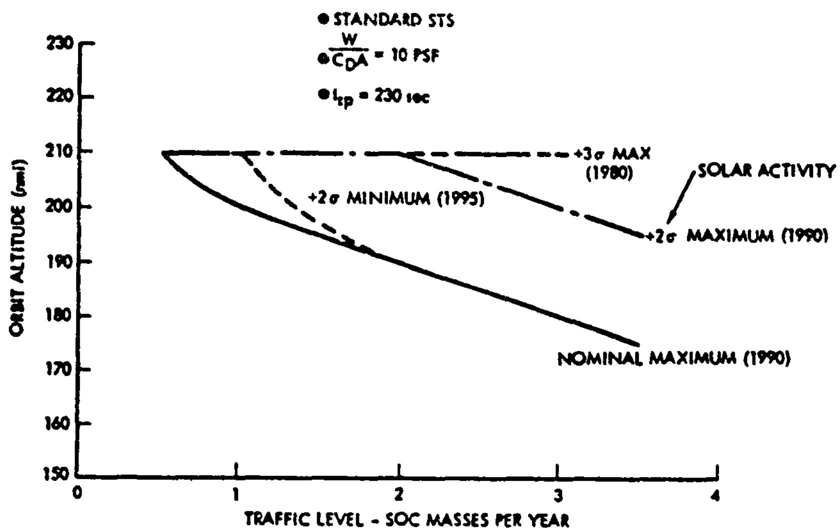

FIGURE 1.22 ATMOSPHERE AND TRAFFIC EFFECTS ON OPTIMUM SOC ORBIT ALTITUDE

Yearly orbit makeup propellant for these optimum logistics altitudes are presented in Figure 1.23. They obviously follow the inverse pattern of the atmospheric effects in altitude shown in the previous figure. As logistics traffic increases, the optimum altitude drops and hence, orbit makeup propellant increases. Although the yearly propellant does increase these are still the optimum operating conditions because of the Shuttle delivery performance effects.

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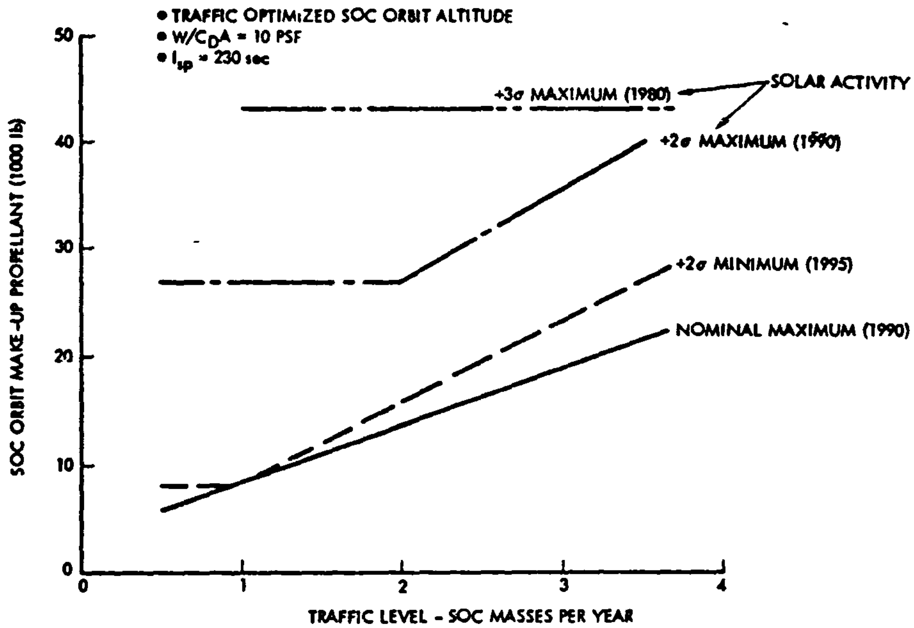

FIGURE 1.23 YEARLY SOC ORBIT MAKEUP PROPELLANT

To develop further insights into logistics performance sensitivities, both incremental and integer Shuttle flight effects were briefly examined. First, to analyze the sensitivity of logistics performance to operations at off-optimum altitudes the incremental increases in the number of Shuttle flights was determined for several traffic levels over a range of orbit altitudes. The results are shown in Figure 1.24. A zero increment is shown for the optimum altitude at each traffic level. As the altitude is varied on either side of this optimum additional Shuttle flights will be required, either to carry more makeup propellant (lower than the optimum altitude) or to account for reduced Shuttle delivery performance (higher than the optimum altitudes).

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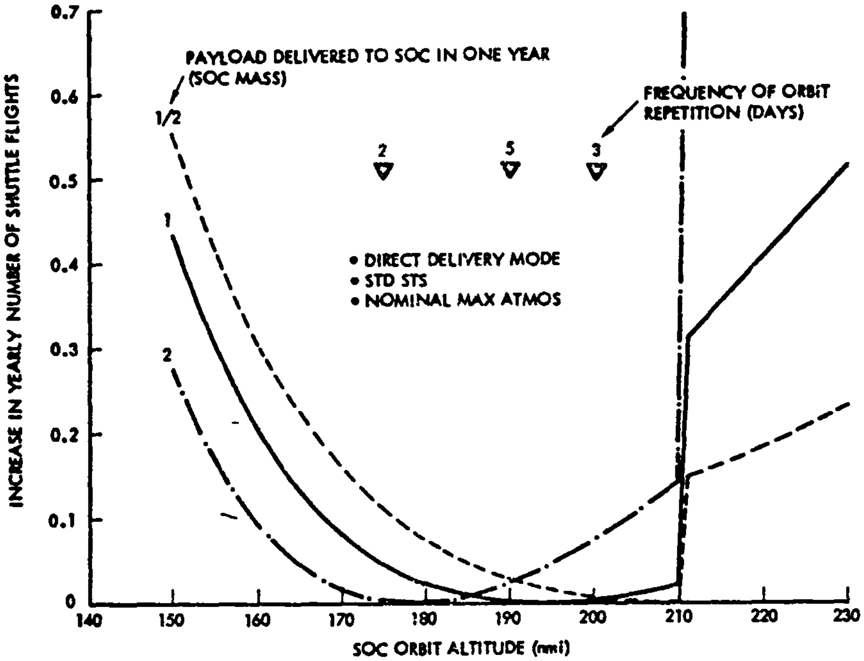

FIGURE 1.24 SHUTTLE FLIGHT SENSITIVITY TO ORBIT ALTITUDE

Sensitivity to off-optimum altitudes is not great. An altitude spread of 50 to 75 Km (30 to 40 n. mi.) is shown to exist for an incremental increase of 0.1 Shuttle flights per year within the range of traffic levels investigated. Perhaps even more surprising is the almost trivial penalty at the 350 Km (190 n.mi) crossover point of the high and low traffic levels. Here, for the full range of traffic the worst penalty increment is 0.025 Shuttle flights per year which represents less than a one percent penalty for the low traffic case (one-half SOC mass per year) and is even a smaller percentage for the higher traffic levels. This altitude is also shown to correspond to a five-day orbit trace repetition pattern. When indexed to KSC this would allow Shuttle launch opportunities direct to the SOC every five days without phasing orbit coast operations which could add a full day or more to the ascent delivery profile. These data would be affected by different atmospheric densities and SOC drag characteristics, but are indicative of the relative low sensitivity to off-optimum altitudes.

This low sensitivity is further exhibited in the integer Shuttle flight plots in Figure 1.25. Here, the incremental Shuttle flight trends in the previous figure were extended to complete integer increments and the actual number of Shuttle flights required to deliver the total SOC cargo is shown. In the preceding figure (Figure 1.24) the optimum altitude solutions for the various traffic levels may call for "n" plus a fraction of Shuttle

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launches. Therefore, the delta Shuttle flight increments were not necessarily referenced to an integer number of Shuttle flights. This effect was accounted for in the data of Figure 1.25 and the integer "jump" points here represent the true flight requirements. As implied in the incremental data, wide variations in SOC operating orbit altitudes are possible before the logistics penalties become large enough to require an additional yearly Shuttle flight. This is because the Shuttle payload capability is quite large compared to makeup propellant requirements.

The main conclusion is that the optimum logistics performance is not highly sensitive to changes in orbit altitude. Thus, a simple strategy algorithm for real time control of SOC operating altitudes to meet the changing configuration, atmosphere, and traffic conditions should suffice.

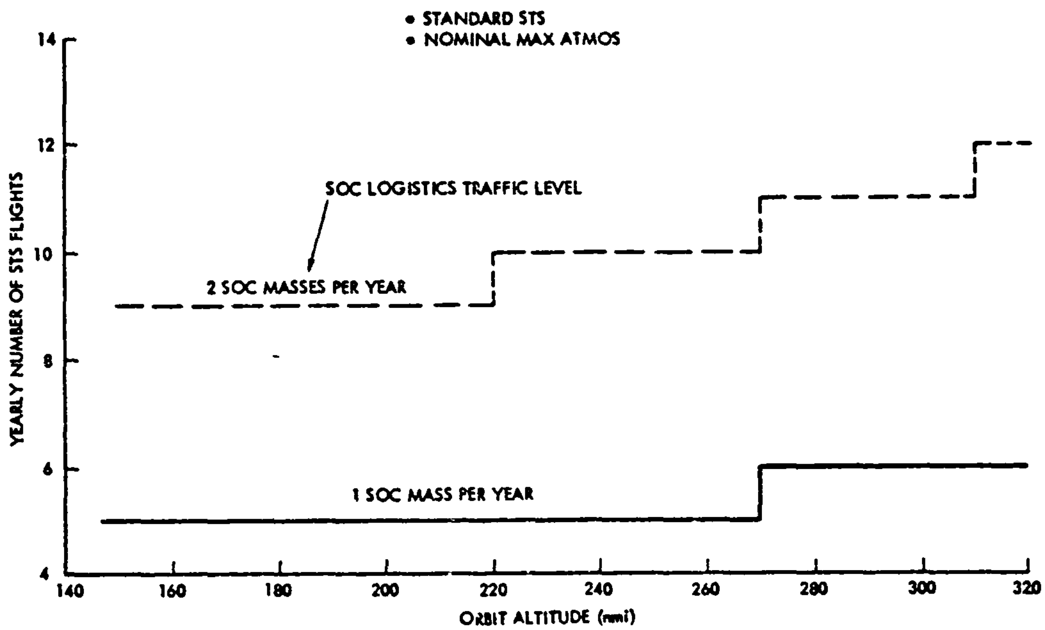

FIGURE 1.25 INTEGER SHUTTLE FLIGHT EFFECTS

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Tug Assisted Delivery Mode

The tug assisted delivery mode used in this analysis is pictured in Figure 1.26 along with the assumed tug characteristics. Tug propellants must be carried in the logistics for this mode as well as all other cargo elements identified for the direct delivery mode. The assumed tug features are representative of a large teleoperator type concept with an advanced hydrazine monopropellant propulsion system. While a full sized space-based tug could also do the SOC assist job, the above concept was selected as a low cost approach which could potentially be available in the same time frame as the SOC.

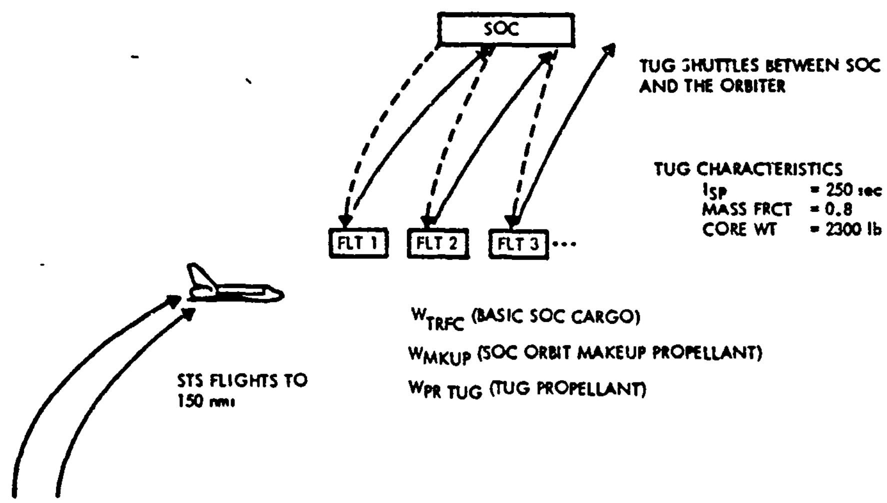

FIGURE 1.26 TUG ASSISTED DELIVERY MODE

The logistics performance for this mode is presented in Figure 1.27. The logistics penalty factor is shown as a function of orbit altitude for traffic levels ranging from one-half to two SOC masses per year. A distinguishing feature of these data is the absence of the OMS kit "barrier". Since the Shuttle only flies to 275 Km (150 n.mi) altitude in this mode, the performance discontinuity associated with its use does not occur. This, combined with the improved staging efficiency from not flying the heavy Shuttle to high altitudes results in optimum logistics altitudes slightly higher than for the direct delivery mode, particularly for those conditions which previously optimized at the OMS barrier altitude (390 Km or 210 n.mi.).

For the tug assisted mode the optimum orbit altitudes are approximately 410 Km (220 n.mi), 370 Km (200 n.mi) and 350 Km (190 n.mi) for traffic levels of one-half, one and two SOC masses per year respectively. The corresponding logistics penalty factors are approximately 1.1, 1.075 and 1.05. These compare with 1.11, 1.09 and 1.06 respectively for the corresponding values with the direct delivery mode, not a large difference between modes in overall logistics performance.

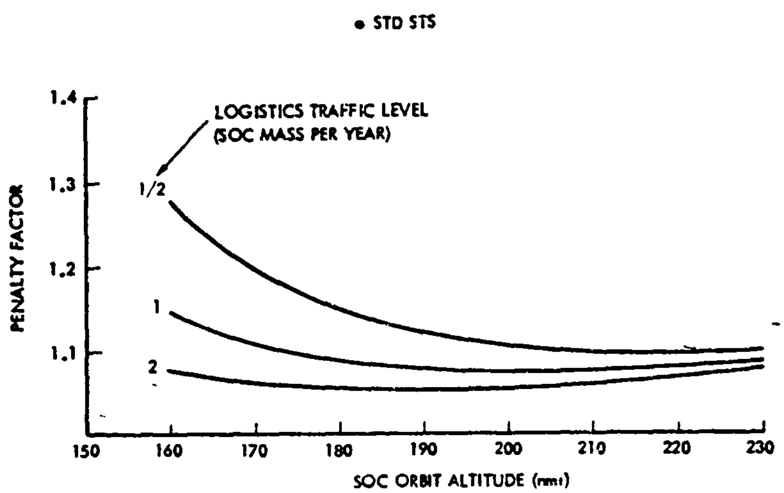

FIGURE 1.27 LOGISTICS PERFORMANCE FOR THE TUG ASSISTED DELIVERY MODE

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The sensitivity of logistics performance to non-optimum altitudes was also briefly examined in this delivery mode, Figure 1.27A. As in the direct delivery case, low sensitivity to off-optimum altitudes is exhibited. Operating altitude ranges exceeding 90 Km (50 n.mi) are possible within a penalty increment of 0.1 Shuttle flights per year.

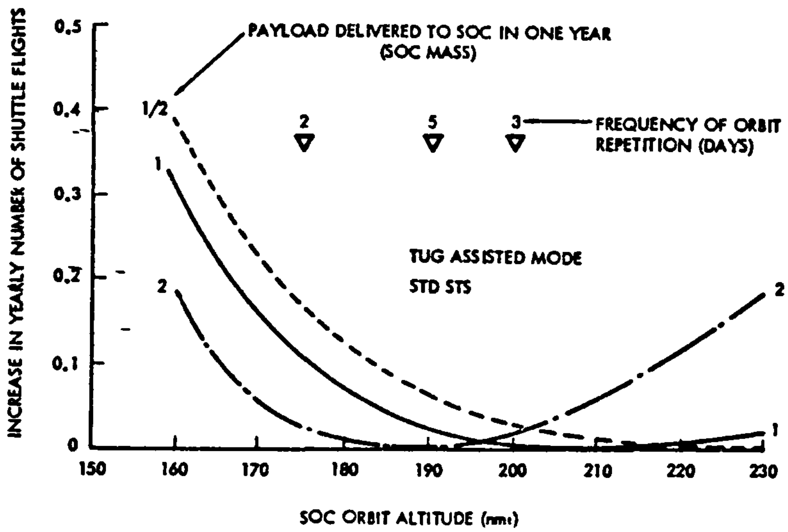

FIGURE 1.27A SHUTTLE FLIGHT SENSITIVITY TO ORBIT ALTITUDE - TUG ASSISTED

<u>OTV Flydown Delivery Mode</u>

This delivery mode is focused on the OTV component of SOC logistics traffic which includes the OTV propellants and payloads to be transported by the OTV to GEO. Other basic SOC resupply cargo is carried direct to the SOC in the orbiter. Due to the "split" cargo operations a number of factors must be established to maintain data consistancy among the modes. First, the OTV concept must be defined. For this analysis an OTV made up of two identical stages was selected and was assumed to be mainly used to deliver payloads to geosynchronous orbit and then return empty (both stages). A baseline OTV flydown profile was then established to account for all of the traffic components and flight segments. This baseline profile is outlined in Table 1.3 along with the standard SOC based OTV mode to highlight the differences. In all cases the OTV propellant sizing calculations are based on the following stage characteristics, Isp = 460 sec and stage propellant fraction = 0.9. The additional ΔVs to fly the OTV down to 275 Km (150 n.mi) from the SOC and then to stage the GEO mission from this altitude instead of the SOC altitude are also included.

TABLE 1.3 BASELINE PROFILE OTV FLYDOWN MODE

MISSIONS	
STD SOC BASED OTV MODE	OTV FLY-DOWN MODE
STS DELIVERS TO SOC • OTV PROPELLANT } BASIC TRAFFIC MANIFEST • GSO PAYLOAD } • SOC ORBIT MAKEUP PROP.	STS DELIVERS TO SOC • SOC ORBIT MAKEUP PROPELLANT STS DELIVERS TO 150 NMI • GSO PAYLOAD (SAME AS IN THE SOC SUPPORTED MISSION) • OTV PROPELLANT TO COMPLETE MISSION
• OTV_1 PERFORMS PART OF TRANSFER MISSION AND RETURNS TO SOC • OTV_2 COMPLETES TRANSFER TO GSO • OTV_2 DELIVERS PAYLOAD • OTV_2 RETURNS EMPTY TO SOC	• OTV DEORBITS FROM SOC ALTITUDE TO MEET STS FLIGHTS AT 150 NMI • OTV_1 PERFORMS PART OF TRANSFER MISSION AND RETURNS TO SOC • OTV_2 COMPLETES TRANSFER TO GSO • OTV_2 DELIVERS PAYLOAD • OTV_2 RETURNS EMPTY TO SOC

The resulting logistics performance is presented in Figure 1.28. The logistics penalty factor is shown as a function of traffic level. SOC altitude was held constant at 370 Km (200 n.mi) for the generation of these data. This greatly reduced the number of iterative calculations required, but still gives a good indication of the performance of this mode. The 370 Km altitude is near the middle of the range of optimum altitudes for a wide spread of conditions. Thus, while slightly improved performance results could be expected with optimum altitudes for this mode, a good basis of comparison with the other modes is provided.

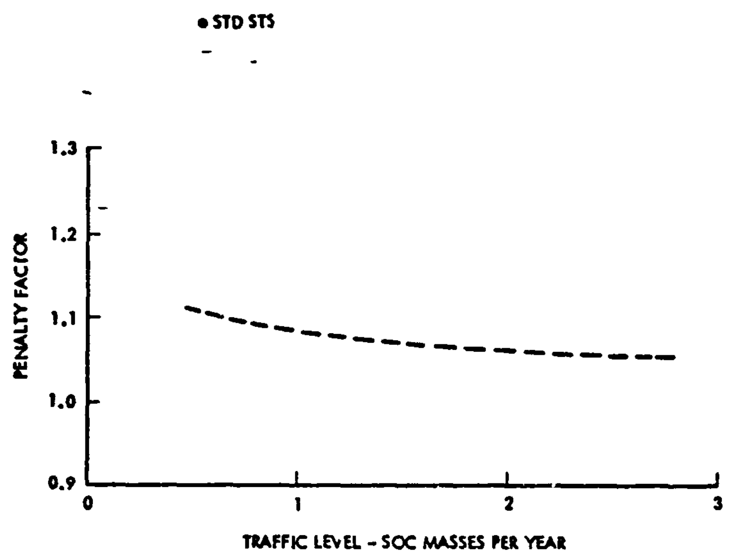

FIGURE 1.28 LOGISTICS PERFORMANCE FOR THE OTV FLYDOWN MODE

Delivery Mode Comparison

Figure 1.29 compares the logistics performance of the three basic delivery modes. The logistics penalty factors for the three modes are shown superimposed as a function of traffic level. The maximum penalty factor spread across all of these modes is an increment of approximately 0.02. For traffic levels of one SOC mass per year and above, this represents less than a two percent difference among the modes. While slight adjustments in the relative placement of the different modes would be introduced by changes in the assumptions on tug and OTV performance, the effects are small. In all modes the Shuttle must carry all components of cargo to at least the 275 Km (150 n.mi) altitude. Differences in stage performance for the small ΔVs required to transfer cargo from this altitude to the 370 or so Km SOC altitude have little effect on the overall logistics performance.

Thus, the direct delivery mode is best. Both the tug assisted delivery mode and the OTV flydown mode require the development of either additional hardware (tug) or more complex operations of the OTV. The direct STS delivery mode, on the other hand, represents the simplest and most straight forward strategy to deliver a variety of payloads to the SOC. It requires just a single ascent rendezvous without the complications of intermediate orbit phasing/plane change considerations and there are no extra payload handling services required for transferring payloads to the other vehicles as well as to the SOC.

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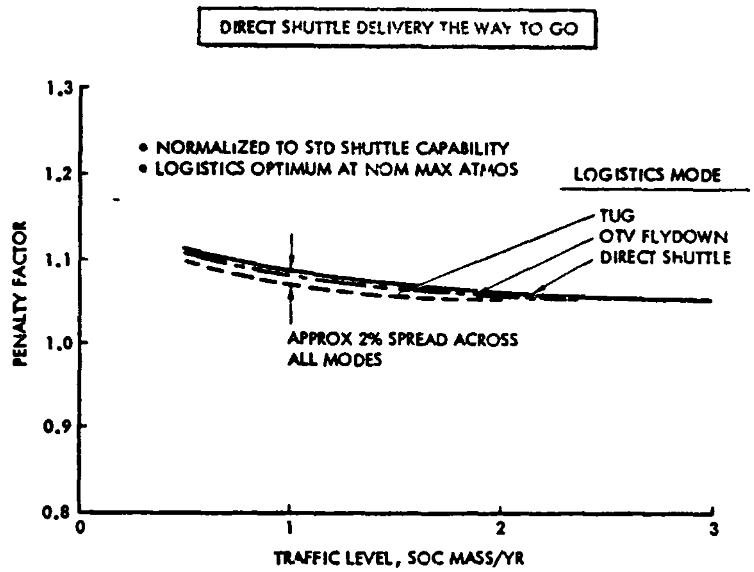

FIGURE 1.29 DELIVERY MODES COMPARISON

Augmented Thrust STS Logistics Performance

The logistics performance of the thrust augmented STS with the direct delivery mode was evaluated to determine the potential effects on SOC orbit altitude. The augmented STS payload capability used in this analysis is shown in previous Figure 1.4 and discussed in Section 1.2. The 29500 Kg (65,000 lb) landing weight limit of the standard STS was not imposed. However, if this limit cannot be exceeded, the logistics driven SOC altitude would correspond to the highest altitude the 29500 Kg payload could be carried, approximately 550 Km (300 n.mi), depending on the amount of down cargo.

The logistics performance for the augmented STS is presented in Figures 1.30 through 1.33. The logistics penalty factor is shown as a function of orbit altitude for logistics traffic levels ranging from one-half to three SOC masses per year. These data are normalized to the augmented STS payload capability at 275 Km (150 n.mi) in the same manner as the previous results were normalized for the standard STS. Thus, the penalty factor here represents the ratio of augmented Shuttle flights to carry the total SOC cargo to each indicated altitude to the number of augmented Shuttle flights required to carry just the basic SOC cargo (no makeup propellant) to 275 Km.

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As it did in the standard STS, the OMS discontinuity again limits the logistics optimum altitudes at the lower traffic levels where makeup propellant tends to dominate. Note here, however, that the OMS kit discontinuity occurs at 370 Km (200 n.mi) instead of the 390 Km (210 n.mi) for the standard STS because of trajectory shaping differences in optimizing the ascent profile.

A comparison of these results with the standard STS logistics performance is presented in Figure 1.34. The logistics penalty factor as a function of altitude is shown for both STS configuration at a representative traffic level of one SOC mass per year along with the optimum logistics altitudes for both as a function of traffic level. The main differences are slightly lower penalty factor values for the augmented STS along with optimum logistics orbit altitudes which are approximately 18 Km (10 n.mi) higher. Both of these effects are caused by the slightly shallower payload drop-off slope with altitude associated with the augmented STS performance. As discussed in Section 1.2, the payload slope for the augmented STS is minus 50 Kg/Km (59 lb per n.mi) altitude compared to minus 63 Kg/Km (75 lb per n.mi) for the standard STS.

Thus, while the augmented STS would be able to meet the SOC logistics requirements with fewer Shuttle flights and possibly at reduced dollars per pound, the currently defined SOC configuration and its supporting logistics requirements are within the capabilities of the standard Shuttle. Therefore, as noted in the figure, the standard Shuttle can do the job; an advanced Shuttle is not a requisite for SOC. However, the variable altitude strategy recommended for the SOC is compatible with the augmented STS and would allow optimum utilization of all STS improvement steps as they become available.

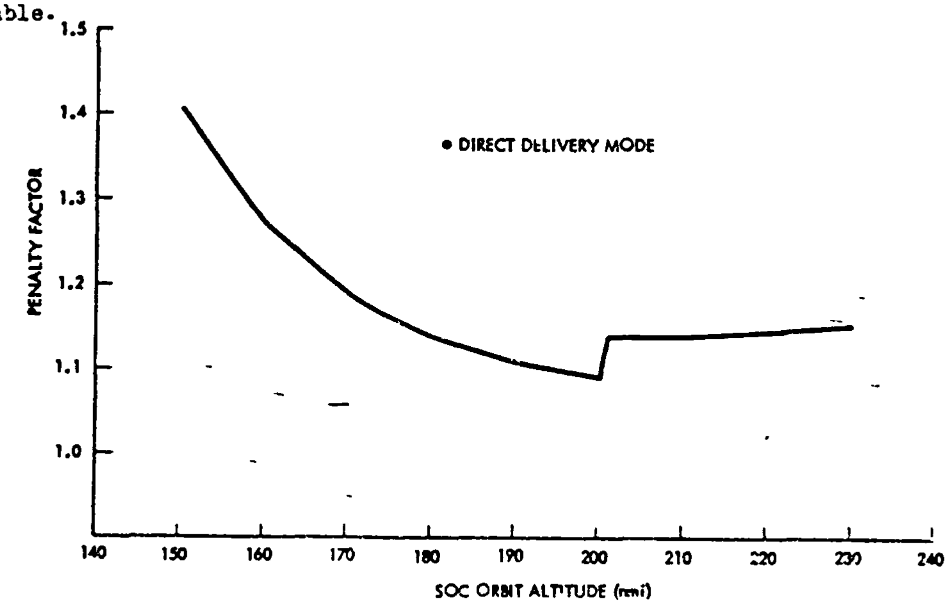

FIGURE 1.30 LOGISTICS PERFORMANCE FOR AUGMENTED STS - ONE HALF SOC MASS PER YEAR

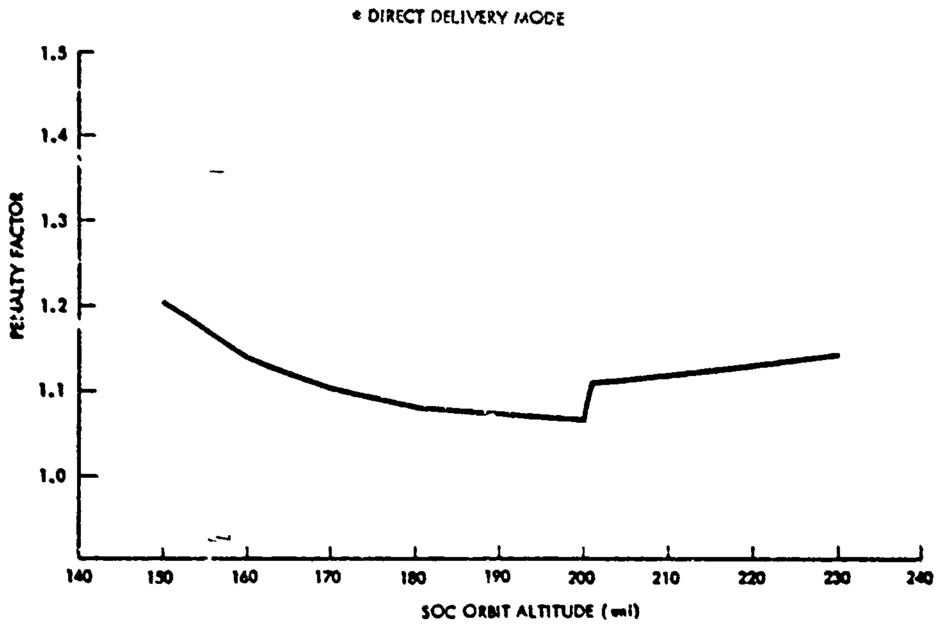

FIGURE 1.31 LOGISTICS PERFORMANCE FOR AUGMENTED STS - ONE SOC MASS PER YEAR

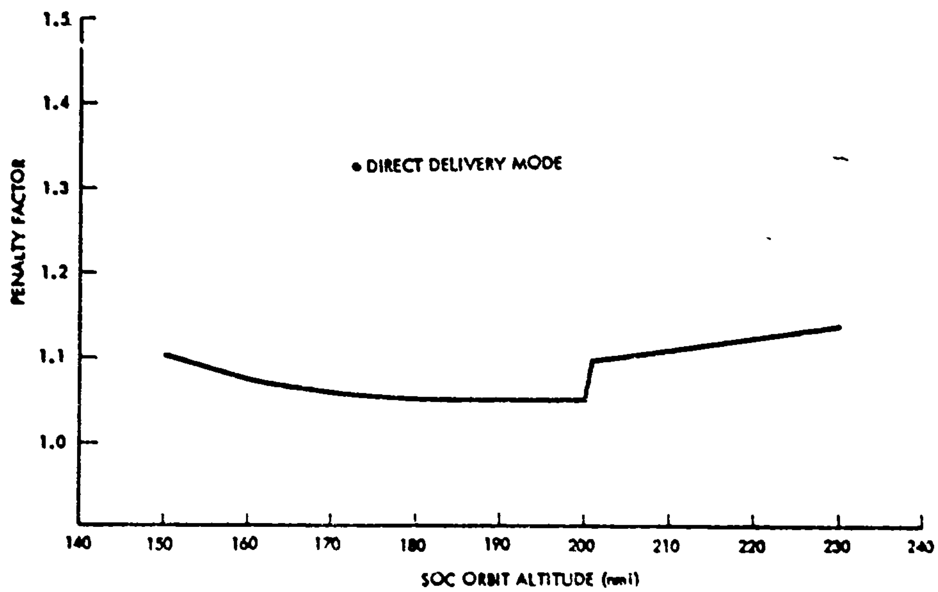

FIGURE 1.32 LOGISTICS PERFORMANCE FOR AUGMENTED STS - TWO SOC MASSES PER YEAR

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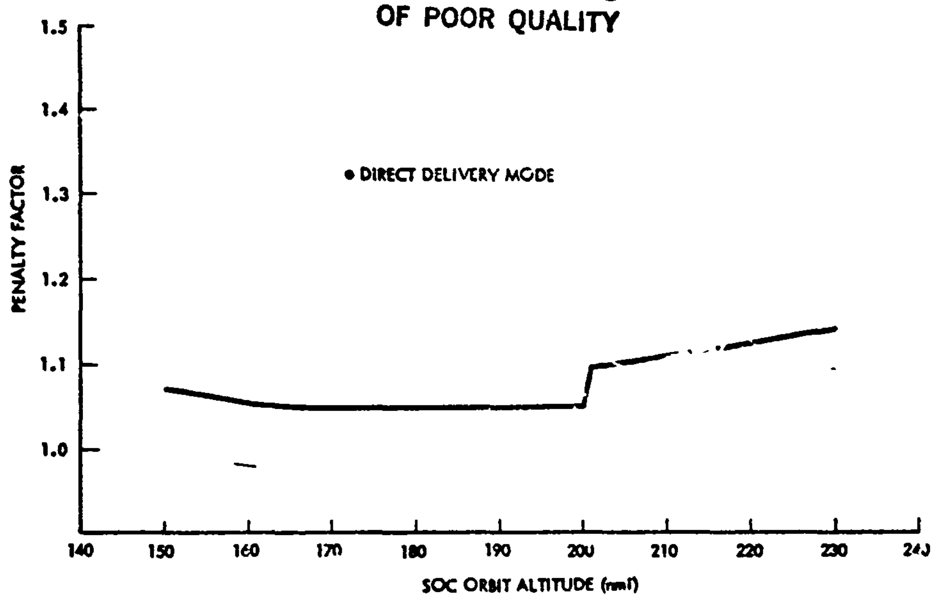

FIGURE 1.33 LOGISTICS PERFORMANCE FOR AUGMENTED STS - THREE SOC MASSES PER YEAR

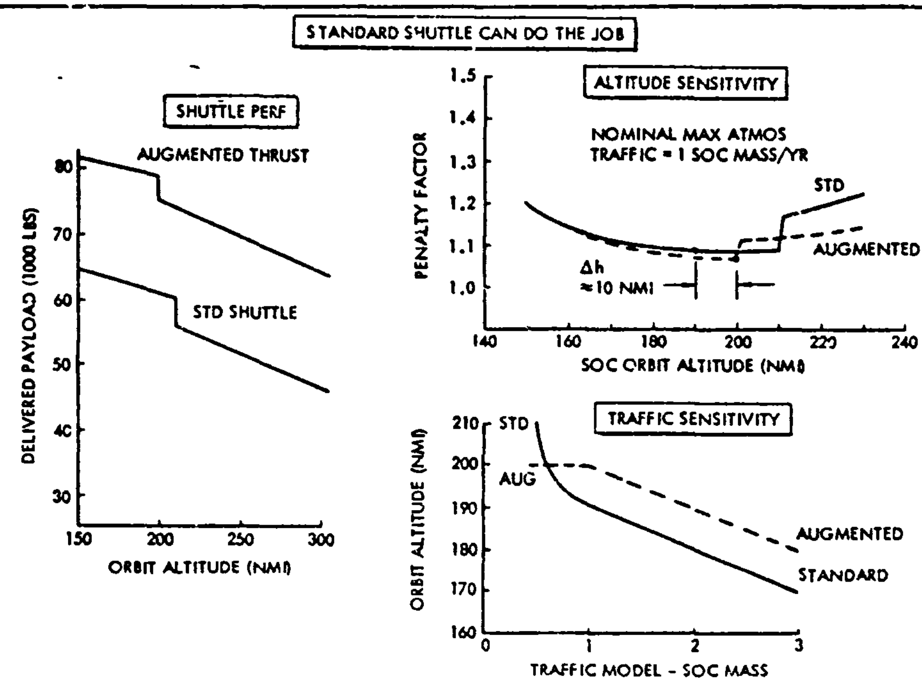

FIGURE 1.34 DELIVERY PERFORMANCE COMPARISON

1.7 SOC ORBIT ALTITUDE STRATEGIES

The principal consideration for choosing a SOC orbit altitude strategy is to always fly the SOC at a mission safe altitude. The issue is: What is a safe altitude? Does it vary? And, what are the cost implications of achieving the desired level of safety?

Orbital safety is usually equated with orbital decay - i.e., orbital lifetime - without any orbit altitude make-up capability. Thus, the criteria for acceptable safety will be the number of days the SOC could remain above a specified orbit under severe emergency conditions. For the purposes of this study the minimum acceptable SOC orbit altitude is assumed to be 185 Km (100 n.mi). From this altitude a catastrophic SOC entry into the Earth's atmosphere would be only a matter of a few days.

It may be noted that this criteria implies not only the total loss of orbit makeup capability, but also the inability of the STS fleet to deliver and perform the necessary maintenance.

A number of temporary actions can be taken to extend the lifetime of a station as large and complex as the SOC. These could include a more "optimum' attitude flight mode and/or a decrease in the exposed solar array area. However, for conservatism these actions were not assumed in the subsequent lifetime analysis.

Three SOC orbit altitude strategies were evolved from the analyses in the previous sections. Two of these involve fixed altitude options, while the third is a variable altitude concept. These three strategies together with some of their pertinent attributes are summarized in Table 1.4.

TABLE 1.4 SOC ORBIT ALTITUDE STRATEGIES

LOGISTICS-DRIVEN CONSTANT ALTITUDE	• PROVIDES GOOD LOGISTICS PERFORMANCE • ALLOWS PREDICTABLY REPEATING GROUND TRACE • SIMPLIFIES MISSION PLANNING • RESULTS IN RELATIVELY SHORT ORBIT LIFE FOR PERIODS OF HIGH ATMOSPHERIC DENSITY
DECAY-DRIVEN CONSTANT ALTITUDE	• PROVIDES ASSURED SAFE ORBIT LIFE • REQUIRES EXTRA SHUTTLE FLIGHTS BECAUSE OF HIGH ALTITUDE DELIVERY • REQUIRES OMS KIT EXCEPT FOR PARTIAL SHUTTLE LOADS
VARIABLE ALTITUDE STRATEGY	• COMBINES SAFETY WITH LOGISTICS EFFICIENCY • ALLOWS FOR UNUSUAL SOLAR ACTIVITY • ALLOWS FOR VARIATIONS IN SOC LOGISTICS TRAFFIC

The logistics driven constant altitude strategy would result in the lowest logistics penalty factor over the long term average for the SOC program life. However, since it would be established for a particular traffic model and for a particular solar activity/atmospheric density condition it would lack flexibility if any of these two items change. Short term penalties would be incurred during periods of high traffic and/or low density. By its constant altitude nature it would present the least variables to the mission planner. The key characteristics for this strategy option are presented in Table 1.5. An additional significant advantage of this strategy is that the resulting SOC altitudes are all below the point where OMS kits would be required by the STS.

TABLE 1.5 LOGISTICS-DRIVEN CONSTANT ALTITUDE STRATEGY (STANDARD STS)

	TRAFFIC MODEL—SOC MASS		
	1/2	1	2
NOMINAL MAXIMUM ATMOSPHERE (1990)			
•OPTIMUM ALTITUDE (NMI)	210	200	190
•6-MO ORBIT MAKEUP PROPELLANT (LB)	6000	8600	12,700
ORBIT LIFETIME—NOM. MAX. ATMOS. (DAYS)	220	155	105
ORBIT LIFETIME— $+2\sigma$ MAX. ATMOS. (DAYS)	70	55	40
LOGISTICS PENALTY FACTOR	1.127	1.099	1.075
$+2\sigma$ MAXIMUM ATMOSPHERE (1990)			
•OPTIMUM ALTITUDE (NMI)	210	210	210
•6-MO ORBIT MAKEUP PROPELLANT (LB)	13,500	13,500	13,500
ORBIT LIFETIME—NOM. MAX. ATMOS. (DAYS)	220	220	220
ORBIT LIFETIME— $+2\sigma$ MAX. ATMOS. (DAYS)	70	70	70
LOGISTICS PENALTY FACTOR	1.311	1.193	1.134

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The principal concern with the logistics driven altitude concept is the orbit lifetime or safety factor. Under the conditions of a $+2\sigma$ maximum (1990) solar activity, orbit lifetimes between 40-70 days may be expected. An even shorter decay life is possible with the more severe $+3\sigma$ design atmosphere.

The major disadvantage of the logistics driven constant altitude strategy could be overcome by positioning the SOC at some higher constant altitude consistent with the desired orbit lifetime criteria and solar activity model. This is called the decay driven constant altitude strategy. The characteristics of two such "safe altitude" approaches are illustrated in Tables 1.6 and 1.7.

TABLE 1.6 DECAY-DRIVEN ALTITUDE STRATEGY (180-DAY DECAY)

	TRAFFIC MODEL—SOC MASS		
	1/2	1	2
NOMINAL MAXIMUM ATMOSPHERE (1990)			
• ORBIT ALTITUDE (NMI)	204	204	204
• 6-MO ORBIT MAKEUP PROPELLANT (LB)	3600	3600	3600
ORBIT LIFETIME: NOM. MAX. ATMOS. (DAYS)	180	180	180
ORBIT LIFETIME: $+2\sigma$ MAX. ATMOS. (DAYS)	60	60	60
LOGISTICS PENALTY FACTOR	1.128	1.099	1.081
$+2\sigma$ MAXIMUM ATMOSPHERE (1990)			
• ORBIT ALTITUDE (NMI)	246	246	246
• 6-MO. ORBIT MAKEUP PROPELLANT (LB)	5500	5500	5500
ORBIT LIFETIME. NOM. MAX. ATMOS. (DAYS)	>500	>500	>500
ORBIT LIFETIME. $+2\sigma$ MAX. ATMOS. (DAYS)	180	180	180
LOGISTICS PENALTY FACTOR	1.361	1.306	1.278

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TABLE 1.7 DECAY-DRIVEN ALTITUDE STRATEGY (365-DAY DECAY)

	TRAFFIC MODEL—SOC MASS		
	1/2	1	2
NOMINAL MAXIMUM ATMOSPHERE (1990)			
•ORBIT ALTITUDE (NMI)	224	224	224
•6-MO. ORBIT MAKEUP PROPELLANT (LB)	1800	1800	1800
ORBIT LIFETIME: NOM. MAX. ATMOS. (DAYS)	365	365	365
ORBIT LIFETIME: +2σ MAX. ATMOS. (DAYS)	100	100	100
LOGISTICS PENALTY FACTOR	1.228	1.210	1.202
2σ MAXIMUM ATMOSPHERE (1990)			
•ORBIT ALTITUDE (NMI)	274	274	274
•6-MO. ORBIT MAKEUP PROPELLANT (LB)	2700	2700	2700
ORBIT LIFETIME: NOM. MAX. ATMOS. (DAYS)	>>500	>>500	>>500
ORBIT LIFETIME: 2σ MAX. ATMOS. (DAYS)	365	365	365
LOGISTICS PENALTY FACTOR	1.385	1.256	1.241

This strategy as the one before suffers from inflexibility. That is, in order to be "absolutely safe" the SOC altitude chosen must conform to the worst solar activity conditions at the peak of the 11 year cycle. This naturally results in unnecessary logistics penalties most of the time when the solar activity is at lower levels.

This lack of flexibility of a constant altitude flight strategy leads to the consideration of a variable altitude approach. With this concept the SOC orbit altitude would be "continuously" or "step wise" changed to conform with new drag environments and/or logistics supply rates.

The decay safe altitude as a function of predicted solar activity is illustrated in Figure 1.35. The altitude varies from 318 Km (172 n.mi) to 494 Km (267 n.mi) depending on the severity of solar activity and the acceptable orbit lifetime.

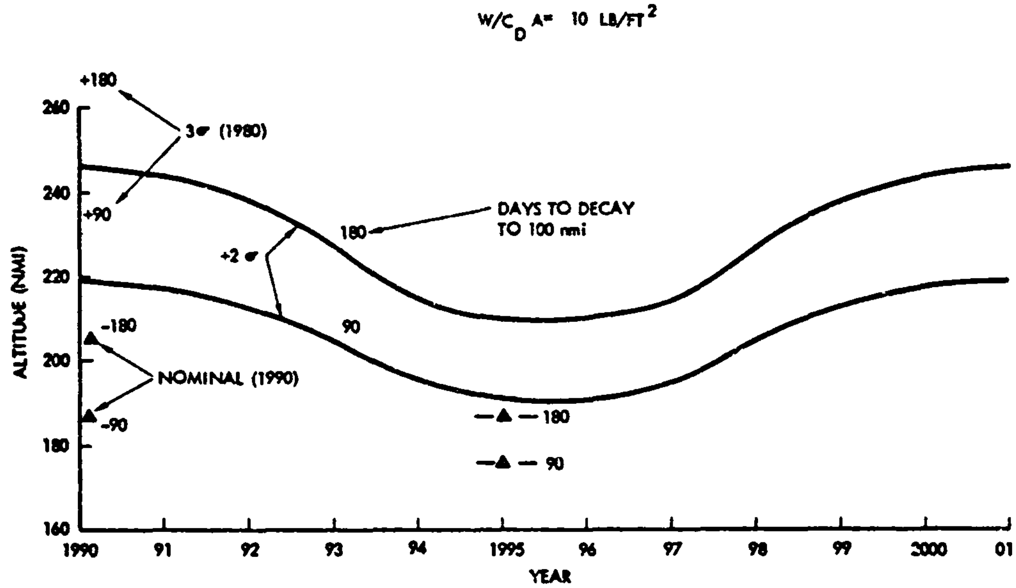

FIGURE 1.35 SOC DECAY SAFE ALTITUDE

For purposes of comparison a 90-day decay period was selected as acceptably safe. This period corresponds to the normal SOC resupply interval and it seems reasonable to assume that most "fixable" emergencies could be met and rectified in 90 days. This 90-day decay safe altitude criteria thus provides one of the driving forces for SOC altitude. The logistics optimum altitude computed for the same traffic and density condition provides the other. The individual altitude histories driven by each of these factors are illustrated in Figures 1.36 through 1.38 for three logistics traffic levels (1, 2, and 3 SOC masses per year). Based on these two limiting choices the SOC orbit altitude will always be at least decay safe (90 days as illustrated) or it will be at the logistics optimum altitude, whichever is higher.

The frequency of altitude adjustments will, however depend on the actual atmospheric fluctations and the particular logistics traffic levels which are experienced. For example, there would be no need to fly the SOC down to the optimum logistics altitude until just before the next logistics flight occurs. This could take advantage of the reduced drag benefits at high altitudes as well as the higher shuttle delivery performance at lower altitudes. Further, the need to change altitude at all could be evaluated for the particular payload manifest on any scheduled resupply mission. Some volume limited payloads may be within the shuttle performance capability without changing the SOC orbit altitude. Thus, a variable altitude strategy permits the operational flexibility to always fly a logistically optimum resupply profile within the constraints of orbital safety.

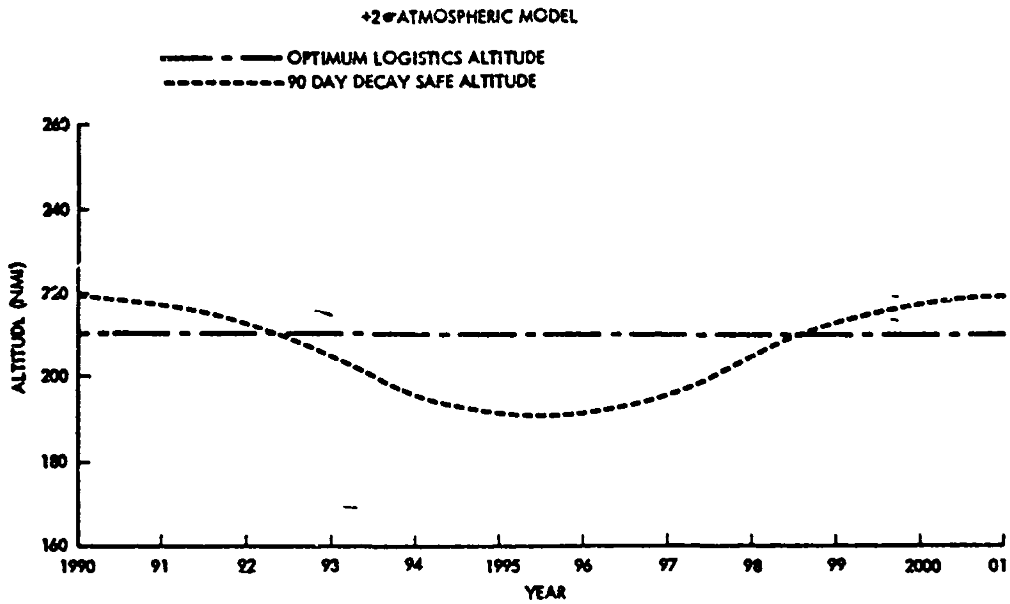

FIGURE 1.36 SOC ALTITUDE FOR ONE SOC MASS PER YEAR TRAFFIC

The data in Figure 1.36 through 1.38 illustrate the effects of traffic levels on the SOC operating altitude. The decay safe altitude for a +2σ atmosphere is shown along with the optimum logistics altitude for each year throughout a typical 11 year solar activity cycle. The logistics optimum altitudes for these data are also based on the drag effects and orbit makeup propellant requirements for the +2σ atmosphere. As shown in Figure 1.36 the optimum logistics altitude is constant throughout the 11-year period for the moderate logistics traffic level of I SOC mass per year (110,000 Kg or 245,000 lb year). This is due to the discontinuity in the STS payload curve at the add OMS kit point (390 Km). This discontinuity (approx. 1360 Kg inert weight) dominates the gradual orbit makeup propellant trends in this altitude range causing the optimum altitude to remain at the add OMS point over a range of atmospheric density conditions. Thus, for the conditions in this figure the SOC would b flown at a decay safe altitude for the period from 1990 to mid-1992 and again from mid-1998 to past 2001. The remaining years, from mid-1992 to mid-1998 would be flown at the optimum logistics attitudes.

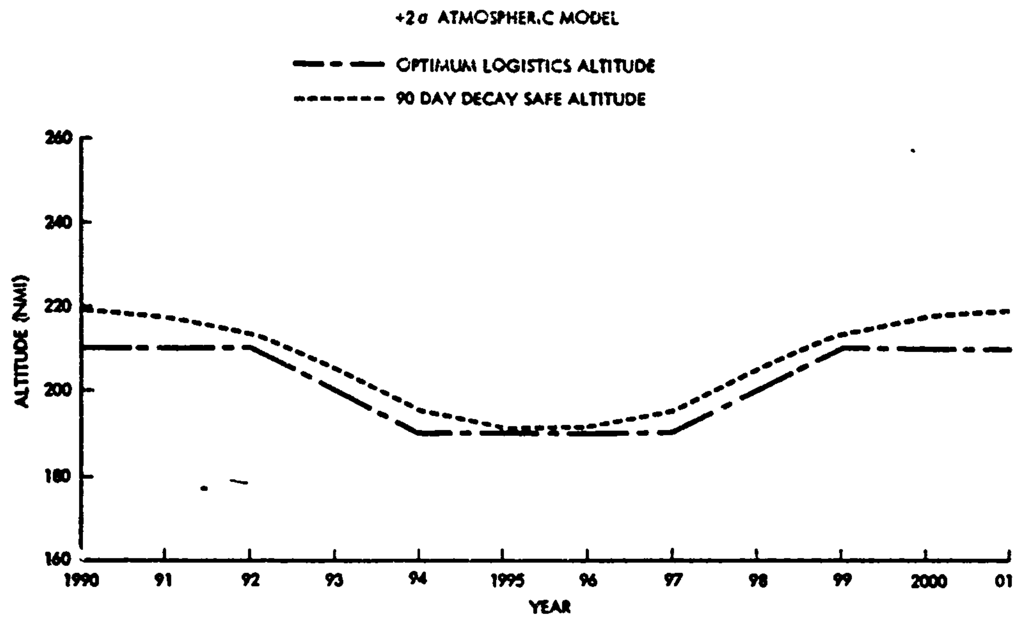

FIGURE 1.37 SOC ALTITUDE FOR TWO SOC MASSES PER YEAR TRAFFIC

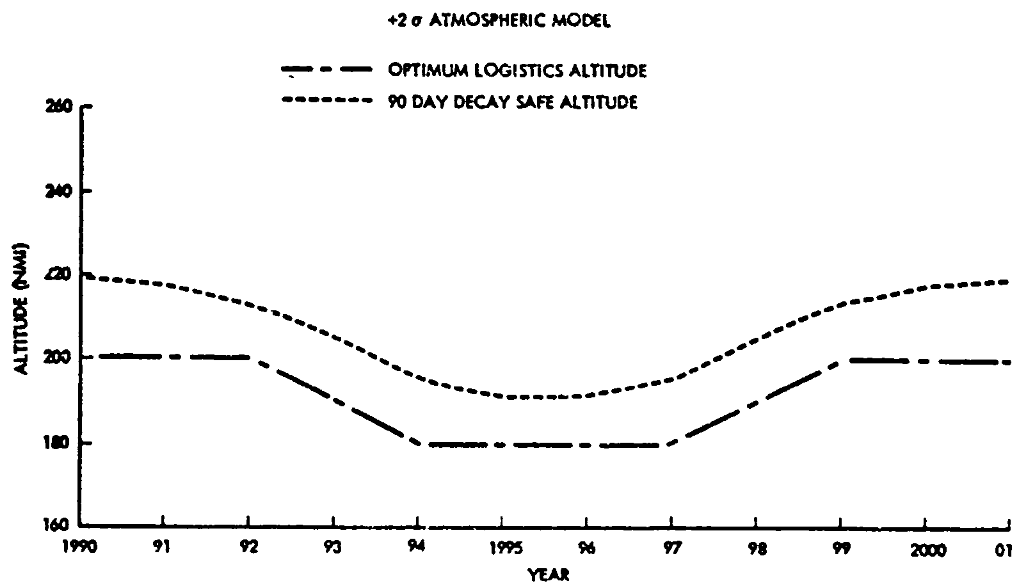

FIGURE 1.38 SOC ALTITUDE FOR THREE SOC MASSES PER YEAR TRAFFIC

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As the traffic levels to SOC are increased the optimum logistics altitude begins to be dominated by the shuttle delivery performance which is better at lower altitudes. Thus, the optimum logistics altitudes in these cases are successively lower for the 2 and 3 SOC masses per year as shown in Figures 1.37 and 1.38 respectively. At 2 SOC masses per year the decay safe and logistics optimum altitudes are quite close over the entire 11 year cycle. However, at the 3 SOC masses per year traffic level the SOC operating altitude would always be dictated by the safe orbit criteria (if the atmosphere remained at the $+2\sigma$ density condition, which is extremely unlikely).

The effect of an even denser upper atmosphere resulting from a $+3\sigma$ 1980 level of solar activity (also specified for STS design) was investigated. The resulting SOC altitudes (logistically optimum and 90-day decay safe) are noted on Figure 1.39 for comparison with the variable $+2\sigma$ atmosphere over the solar cycle period of 1990-2001. Based on this $+3\sigma$ maximum density condition the 90-day decay safe altitude is 236 n.mi. This altitude would be maintained over the entire 11 year solar cycle for the constant altitude strategy case. To always fly SOC at this altitude would unnecessarily penalize the logistics resupply flights during periods of lower solar activity.

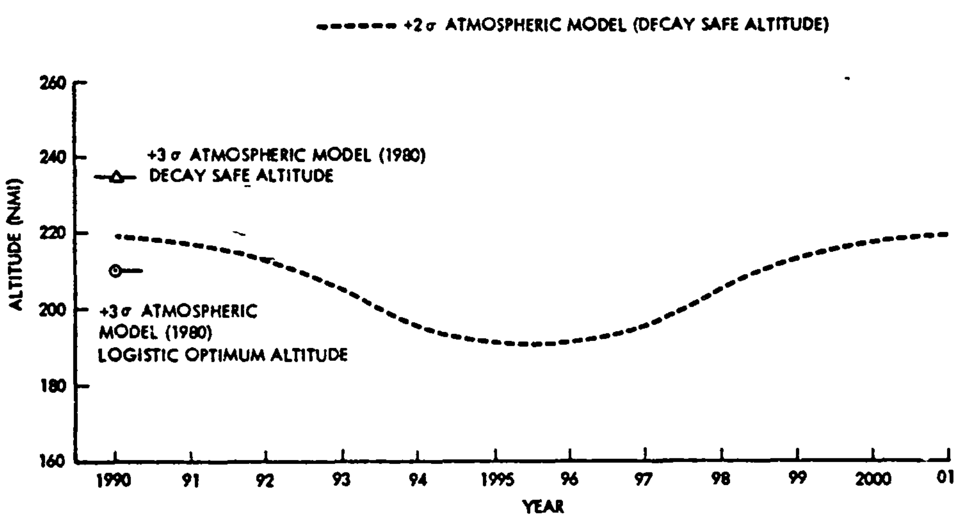

FIGURE 1.39 SOC ALTITUDE FOR ELEVEN YEAR SOLAR CYCLE

Even if the solar activity factor remained statistically high for several years, there would still be a large variation throughout the 11 year cycle. Thus, a variable altitude strategy just allowing for these natural atmospheric variations would save logistics costs. However, even greater savings can be attained by flying at logistics optimum altitudes. The logistic optimum altitude can be expected to be the governing factor over decay safety most of the time. Table 1.8 shows the decay safe and optimum logistics altitudes over the full-range of expected atmospheric conditions during a typical solar cycle. These include the nominal maximum atmosphere which is the expected density at the period of peak solar activity and the nominal minimum which is the expected density during the period of minimum solar activity.

TABLE 1.8 OPTIMUM LOGISTICS ALTITUDES FOR EXPECTED ATMOSPHERIC DENSITY

90-DAY DECAY ALTITUDE	OPTIMUM LOGISTICS ALTITUDE (NMI)		
	1/2 SOC MASS PER YR	1 SOC MASS PER YR	2 SOC MASS PER YR
NOMINAL MAXIMUM ATMOSPHERE h = 185 NMI	210	200	190
NOMINAL MINIMUM ATMOSPHERE h = 172 NMI	200	180	170

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In all cases shown, except for the high traffic case (2 SOC masses per year) with the nominal minimum atmosphere, the optimum altitude is above the decay safe values. Even the high traffic exception is very close to the decay limit. Thus, the full potential logistics savings with the variable altitude strategy is attainable most of the time.

The typical range of savings in SOC logistics costs attainable with the variable altitude strategy is summarized in Figure 1.40. By "flying" SOC at the lowest safe altitude for the prevailing traffic and atmospheric conditions significant savings in logistics flights can be attained by taking advantage of the greater payload delivery capability of the Shuttle to the lower orbit altitudes.

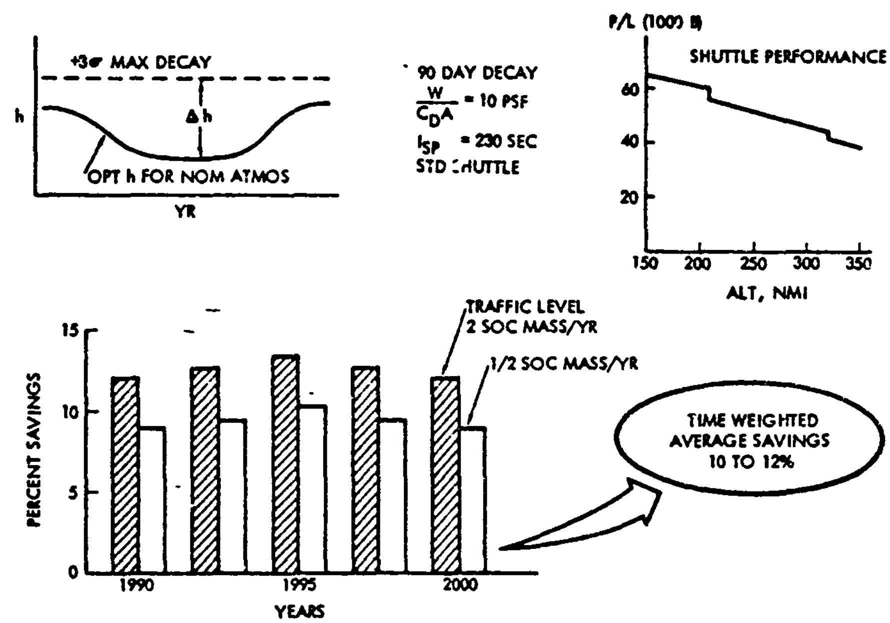

FIGURE 1.40 LOGISTICS SAVINGS WITH SOC VARIABLE ALTITUDE STRATEGY

For example, as shown in the upper right sector of the Figure, the shuttle can deliver approximately 50,000 lbs to 236 n.mi. altitude (the +3σ atmosphere, 90-day decay criteria altitude). In metric units this is 22700 Kg to 435 Km altitude. This would be the lowest altitude permissible for operating SOC with a constant altitude strategy. Payloads above 27200 Kg (60,000 lbs) are attainable at altitudes of 370 Km (200 n.mi.) and below, which is the typical operating range for the variable altitude strategy. This is more than a 20 percent increase in payload capability which can be translated into 20 percent fewer logistics flights to handle the required logistics traffic. Even accounting for the increased SOC orbit makeup propellant required at the lower altitudes, the net savings in logistics flights is from 10 to 12 percent over the full-range of atmsopheric variations throughout a typical 11-year solar cycle.

Thus, a variable altitude strategy is recommended for the SOC.

2.0 BERTHING & DOCKING

2.1 SUMMARY

The objective of Task 2.0 is to explore the SOC/Shuttle interactions related to berthing and docking in order to (1) assess the suitability of the orbiter design for performing these types of operations in conjunction with a large orbiting system, (2) identify key safety related issues, (3) drive out any special SOC requirements introduced by the berthing or docking processes, and (4) analyze the orbiter-SOC docking interfaces for potential commonality with other user programs.

Toward this end, the ability of the orbiter to safely perform terminal closure and docking with SOC was investigated including a number of factors affecting trajectory accuracy, major plume impingement effects and RCS jet failure implications. The key findings from this investigation are underlined in the following paragraphs and accompanied by brief substantiating text.

The Orbiter Can Dock With The SOC

The orbiter flight control system is capable of translation and rotational control accuracies of $\pm$0.015MPS ($\pm$0.05FPS) and $\pm$0.2 deg per sec respectively which are well within the limiting docking contact conditions specified for SOC. Also, there is sufficient thrust/control authority to counter the gravity gradient, aero and other disturbance forces and torques encountered during the docking maneuver sequence. Thus, under normal conditions the orbiter can safely dock with the SOC.

A Runaway Jet Poses A Serious Problem But Appears To Be Controllable With The "Hi-Z" RCS Thrusting Mode

The critical concern with a runaway jet is the case just before docking contact. Here, there is some danger of orbiter-SOC contact outside of the docking envelope. Deviations caused by a runaway jet could drive the orbiter outside of the docking envelope before an abort maneuver to reverse the closing velocity could be completed. However, with the "Hi-Z" thrusting mode the stopping/turnaround times are sufficiently small that deviations from any single runaway jet will always be within the safe docking envelope if the separation distance is within the turnaround envelope at the time of the jet failure. Under these conditions docking will occur with the failed jet still firing. For failures at separation distances greater than this value, safe turnaround aborts without orbiter-SOC contact can be made. Man-in-the-loop simulations with visual cues and system response characteristics are required to confirm this preliminary finding.

A Failed "Off" Jet Is Not A Serious Problem

There is sufficient redundancy in the primary RCS system that no basic thrust authority is lost with any single failed off jet. Hence, all maneuvers can be achieved without degradation. The only effect will be to introduce a 0.24 sec time delay (3 DAP sampling intervals, 80 milliseconds each) in the first thrusting action after the failure. The digital autopilot (DAP) switches to the priority 2 jet in the affected jet group and all thrusting actions thereafter call for the use of this jet and no further 0.24 sec time delays occur.

The SOC Should Be Designed For The Orbiter Docked Condition With Any Single RCS Jet In A Runaway Firing Condition

As indicated above, if the runaway jet occurs just before docking contact is made a more or less normal docking hookup will be made with the jet still firing. If the runaway jet occurs anytime after this docking commitment point is reached up to the time the orbiter flight control system is disabled and powered down following normal docking the docked runaway jet condition is possible. It is estimated that up to one minute may be required to identify and shutdown a runaway jet. Thus, the recommendation to design the SOC for this condition is made.

The SOC Should Be Designed For The Plume Impingement Environment Associated With The "Hi-Z" Thrusting Mode

If a runaway jet occurs at any time in the docking maneuver sequence up to the docking commit point (last few inches) the safest procedure will be to perform a "Hi-Z" abort maneuver. The "Hi-Z" thrusting mode minimizes the turnaround time and distance and precludes orbiter-SOC contact outside the safe docking envelope. Thus, although there is a very low probability of occurance for a runaway jet, particularly during the relatively short time interval (10 to 15 minutes) where the orbiter is relatively close to the SOC, it can happen, and the SOC design should be capable of tolerating the resulting plume environment (9 +Z thrusters and possibly pulsed firings of X and/or Y thrusters).

RCS Plume Effects From Normal Docking Operations are Relatively Mild

The orbiter $+Z_B$ thrusters tend to receive little use during final closure operations for normal docking. Most thruster actions will be for corrections in orbiter X and Y body directions. Almost all close in thruster action for normal docking will be minimum impulse adjustments, approximately 80 milliseconds in duration. These brief bursts will be mostly single or dual thruster firings and will inherently be aimed away from the central SOC modules by the nature of the thruster geometry. What little impingement exists in these cases will be mostly from the very low flux region of the plume field. Thus, the main concern for normal docking is the cumulative effects of mass deposition on sensitive SOC surfaces. These mass depositcs will buildup over the years with the many repeated dockings required. These effects require further, detailed study.

Special SOC Provisions Are Probably Required For Protection From Hi-Z Abort Thrusting Plumes

Although close in aborts during docking are a very low probability event, the combined effects of 9 Z-thrusters firing directly on the SOC for a period of two seconds or more can be severe. Large quantities of exhaust products can he quickly deposited on the frontal surfaces exposed to the plume. Large control disturbances can be induced by plume impingement forces. For the baseline SOC configuration pitching moments exceeding 75000 ft lb and yawing moments approaching 20,000 ft lb are possible. Shielding is probably required for OTVs parked on the flight support and servicing facility beneath the service modules. Plume induced rippling and shearing forces on the delicate MLI thermal protection blankets can cause extensive damage. Thus, significant attention must be given in the SOC design for protection against these severe plume induced environments.

A Standard Mating Interface Can Be Provided

The principal output of the docking/berthing design concept was the development of a standardized mating interface that would accommodate the SOC module mating, the orbiter to SOC mating, and also be compatible with other programs requiring the mating of modules/pallets, and interfacing with the orbiter. A standard mechanical alignment and latching concept was developed as well as a standard utilities interface arrangement. The utilities arrangement dedicates specific areas for the various utilities crossing the interface, i.e., electrical power, data, air distribution, etc. The standard interface also provides a 1 meter clear opening which will accommodate crew transfer through the interface for either a suited crewman or a shirtsleeve crewman.

All of the utilities are remotely actuated in order to make the interface connections. These connections are made after the mechanical mating has been accomplished and verified. The remote actuation mechanism has a manual override capability. Sufficient volume is available to permit either a shirtsleeve or suited crewman to perform maintenance operations within the mated interface.

The mechanical and alignment interface can either be an active element which will accommodate a docking maneuver with an active attenuation system, or be a passive element which will accommodate a berthing mate without active attenuation.

A Docking Module Compatible With the Orbiter and Accommodating the Standard Mating Interface is Feasible

A docking module concept was developed that incorporated the standard mating interface. The docking module provides a pressurized, shirtsleeve, transfer capability between the orbiter and the SOC or any other pressurized spacecraft. The standard utilities interface arrangement, developed for the SOC modules interface, provides the utilities across the docking module interface.

The docking module also interfaces with the existing space lab tunnel adapter, but is structurally supported independently by the payload bridge fitting structure and latches. Isolation of the docking loads from the tunnel adapter is accomplished with the use of a flexible seal member. The docking module has the capability to extend .38M (15") above the mold line of the orbiter and when retracted provides a .9M (36") clearance below the payload bay doors. This excursion permits docking clearance and EVA clearance for payload bay contingencies. A 1 meter clear opening is maintained throughout the module.

Docking Module is not Required to be an Airlock, But can be Provided if Growth Applications Warrant

By utilizing the standard orbiter crew cabin arrangement, which includes an airlock inside the cabin, no additional airlock capabilities are required. Any EVA activities can be accomplished through the onboard airlock. However, if, for instance, a space lab requires the capability for EVA while retaining shirtsleeve passage between the crew cabin and the space lab, the docking module can become an airlock. The installation of a hatch in the upper end of the docking module is the principal change that would be necessary in order to provide the airlock capability.

Docking Module Acceptance for Unmanned Systems

Even though the docking module is designed for pressurized activities, it is capable of being mated to unmanned, unpressurized spacecraft. The utilities connections across the interface are still applicable and will permit EVA maintenance if required.

EVA passage through the docking module, however, requires that the mating spacecraft provide clearance for a suited EVA crewman to exit the docking module.

The docking mechanism, that portion that contains the guidance, latching, attenuation, and utilities interfaces, can be mounted to other docking or berthing devices that may be more amenable to unmanned activities.

In summary, a standard mating interface was developed that will accommodate the mating of SOC modules, can be used to mate the orbiter to the SOC through a pressurized docking module, and can be utilized in unmanned or unpressurized spacecraft mating.

RMS Berthing Requires Software Mods, But Appears Feasible

High fidelity simulation runs by SPAR have shown the RMS has the basic capability to handle the large masses and inertias associated with orbiter/SOC berthing operations. However, minor changes will be required in the control software to permit stable control of the arm with these large system masses.

Residual motions between the orbiter and the SOC were successfully arrested using a slightly modified manual augmented mode. This mode eliminates the automatic switchover to the joint position hold mode when joint rates are reduced to threshold limits built into the software. Use of normal arm stopping modes which include the automatic switchover feature resulted in undamped oscillations. Since all of the RMS control modes for moving payloads have the builtin switchover feature, it is not possible to reposition the orbiter to the SOC berthing port with the present software. Undamped oscillations would result. However, the same modified manual augmented mode of operation used in the stopping action above produced stable control, thereby permitting the use of the arm for maneuvering the orbiter to the SOC berthing port. Thus, it is felt that the RMS fundamentally has the capability for berthing the orbiter to the SOC, but minor software mods are required for stable control. Additional simulation analyses are required to confirm this finding with the addition of SOC/Orbiter body flexibility effects which were not simulated in the current analysis.

Using the RMS to Berth the Orbiter to the SOC is Feasible, But Requires Minor Software Changes

To minimize drift rates in the joint control electronics, the current RMS control software is configured to automatically switch to a "position hold" (PH) submode when all joint rates are within a 0.05 deg/sec rate threshold. This increases the control gains and results in marginal instability for the large system masses associated with the SOC and the fully loaded orbiter. Simulation attempts to arrest 0.1 fps residual motions between the orbiter and the SOC resulted in undamped oscillations after 800 seconds. Elimination of the switchover to the PH submode produced stable control; motion was well damped within 400 seconds.

An additional software mode is required to automatically maneuver the orbiter to the SOC berthing port after the residual motions are nulled. The current arm maneuvering software utilizes a "washout zone" to smooth the stopping action at the designated position. Within the washout zone, distance and angle to go data are calculated and fed into the arm control network to insure a smooth transition to the desired final position and orientation. This feedback function has the same effect as the PH submode above, the effective control gains are increased and instability results.

Modifying the arm software to eliminate these instabilities, judged by SPAR to be minor changes, will result in satisfactory berthing performance. Stopping distance and angles were within 18 inches and 5 degrees even for the severe residual motions assumed for the simulation analyses. Wide separation margins between the orbiter and the SOC were maintained and peak arm loads while high were within design limits. Thus, RMS berthing the orbiter to the SOC is deemed feasible, but requires minor modifications to the arm control software.

2.2 PROXIMITY OPERATIONS AND DOCKING

This section presents the detailed analyses of orbiter proximity operations for direct docking with the SOC. Factors influencing closing trajectory accuracy are discussed along with on-orbit flight modes, jet failure effects and plume impingement considerations. The main objective is to establish the general feasibility of docking the orbiter to a large orbiting space system such as SOC and to drive out the major requirements imposed by proximity flight and docking operations.

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2.2.1 The Terminal Closure Problem

Most of the important factors affecting the closing trajectory for docking are shown symbolically in Figure 2.1. The nature of these factors and their significant interactions are briefly described in the following paragraphs to provide a general "feel" for the overall docking problem.

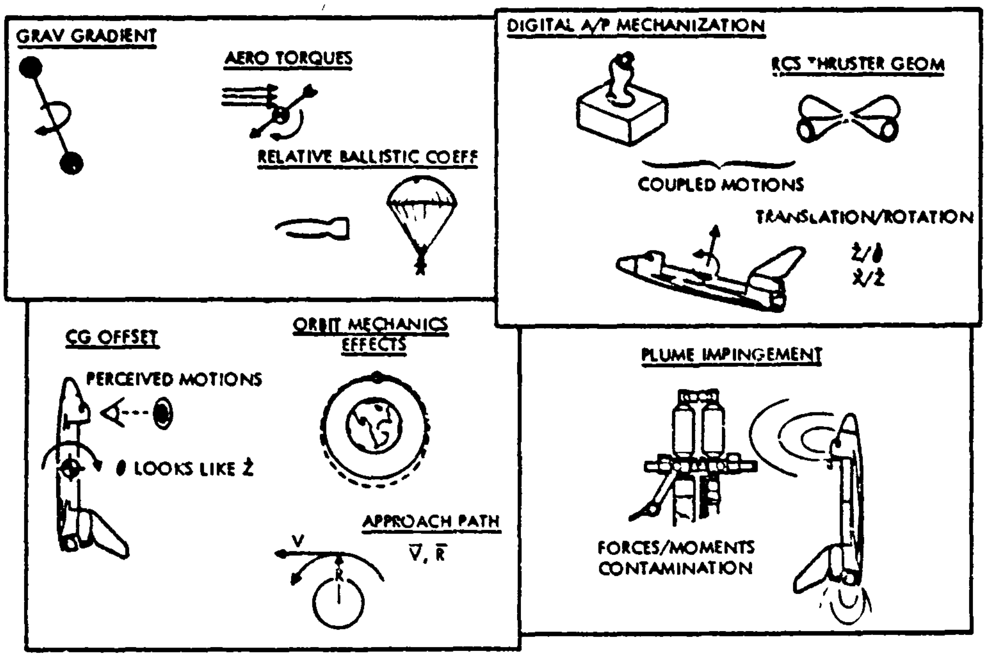

FIGURE 2.1 PROXIMITY OPERATIONS FACTORS

Gravity gradient and aero torques which act on the orbiter vary with orbiter attitude. These attitude related disturbances couple with the orbiter flight control system to produce small trajectory deviations. The RCS thruster geometry in combination with the orbiter digital autopilot (DAP) mechanization produce slightly coupled motions, i.e., translation thrusting produces some rotational motion and rotational thrusting produces some translational motion. Thus, as the orbiter autopilot attempts to correct for the disturbance torques with rotational thrusting pulses, small translational perturbations are introduced. The magnitude of these effects is dependent upon the orbiter attitude and is also affected by variations in approach path direction.

The relative ballistic coefficient (W/C_DA) between the orbiter and the SOC will influence the tendency of the two vehicles to drift toward or away from each other during the closing process. Additional translation control impulses may be required to counter this drift tendency thereby adding to the complexity of the closing operations. Precision control of these translation maneuvers is further complicated by the minimum impulse characteristics of the orbiter flight control system. Each time an RCS thruster is fired it must be "on" for at least 80 milliseconds. The resulting ΔV imparted to the orbiter may be larger than the desired corrective impulse, particularly during the final close in phases of the docking sequence where very fine corrections are desired. This may call for counter corrections and so on, all adding to the piloting burden.

Orbit mechanics effects combine with various approach paths to require differing piloting techniques for each path. Orbit mechanics effects for out of plane approaches (H-bar, along the orbit momentum vector) cause increases in closing velocity as the target is approached whereas for approaches along the orbit radius vector (R-bar) the closing velocity component is reduced by orbit mechanics effects as the target is neared. Further, there are forward or aft drift tendencies with R-bar approaches depending upon approach direction, from below or from above. For approaches along the velocity vector (V-bar) there are drift tendencies to rise or drop depending upon whether the approach is made from the rear or the front. Approaches from skewed directions would have combinations of these effects. As a result. pilot control actions and techniques must be varied for each approach path direction.

The magnitudes of these effects are further influenced by the nominal closing rate selected, i.e., how long the orbit mechanics effects have to take effect and by orbit altitude which changes the aero and gravity gradient environments. The dominance and/or true significance of any or all of these factors can only be fully evaluated through rigorous real-time man-in-the-loop simulations.

Center of gravity (c.g.) offsets introduce two kinds of complexities into the terminal closure process. First, for the orbiter alone, the visual reference cues used by the pilot are affected by the distance between the pilot location and the orbiter c.g. because of ambiguities in the perceived motions. To the pilot, pitch and yaw rotational drifts appear exactly the same as linear motions along the Z and Y body axes respectively, thus compounding the approach control problem. The second complexity introduced by c.g. offsets is related to orbit mechanics effects. If, when the docking ports of the SOC and the orbiter are coaligned, there is a significant offset in their respective c.g.s' orbit mechanics effects will cause small perturbations in the closure path. This is because the two vehicles are in slightly different orbits thereby introducing small relative velocity components. Both in-plane and out-of-plane perturbations can be produced depending upon the c.g. offset direction.

Plume impingement from proximity RCS firings can also influence the precision of the terminal closure operation. Plume impingement forces can cause small SOC orbital velocity changes and attitude disturbances which move its "docking target" and thereby complicate the docking maneuver process. These plume induced forces and moments also cause special loadings and stresses in SOC structural members as well as long term contamination potential.

Thus, there are a number of potentially significant factors which can influence the ability of the orbiter to safely dock with the SOC. The following analyses are aimed at determining their relative significance and the degree to which they can affect docking feasibility.

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2.2.2 Orbiter Flight Control System Description

This section presents the main features and operating characteristics of the orbiter flight control system which are important to the terminal closure and docking operations. Figure 2.2 illustrates a functional schematic of the overall orbiter guidance, navigation and control subsystem. The main elements involved with on-orbit operations are asterisked. These are mostly the sensors, controls and displays, the on-board computer (and DAP software) and the RCS thrusters along with their related signal control devices. There are many complex interactions involving the flight control system which can affect the general docking problem, but the principal concerns here are centered on the RCS thruster locations and geometry and the performance capabilities of the digital autopilot (DAP) controlled flight modes. Therefore, the following discussion is focused on these topics.

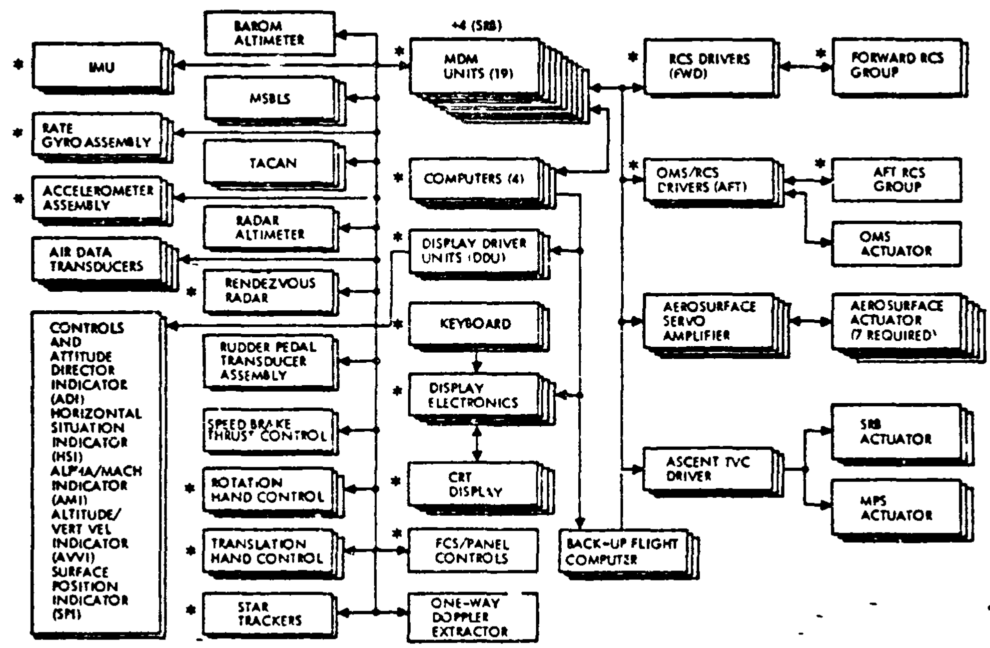

FIGURE 2.2 ORBITER GN&C SUBSYSTEM FUNCTIONAL BLOCK DIAGRAM

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RCS Description

The reaction control subsystem employs 38 primary thrusters and six vernier thrusters to provide attitude control and three-axis translation during on-orbit and other flight phases. The RCS consists of three propulsion units, one in the forward module and one in each of the aft propulsion pods. The basic physical arrangements of these units are illustrated in Figure 2.3 along with basic thruster performance data. Figure 2.4 gives the individual thruster identification nomenclature and shows the general pattern of thruster locations. The 38 main thrusters are clustered into 14 groups according to their thrusting direction. From two to four thrusters are aimed in each of the 14 directions. With the appropriate input commands they can produce three axis rotations and/or translations. Table 2.1 lists the individual thrust components and the locations of thrust application for each thruster. These RCS physical and performance data are used in conjunction with the DAP flight modes and typical orbiter mass properties to analyze key terminal closure issues involving trajectory accuracy and jet failures.

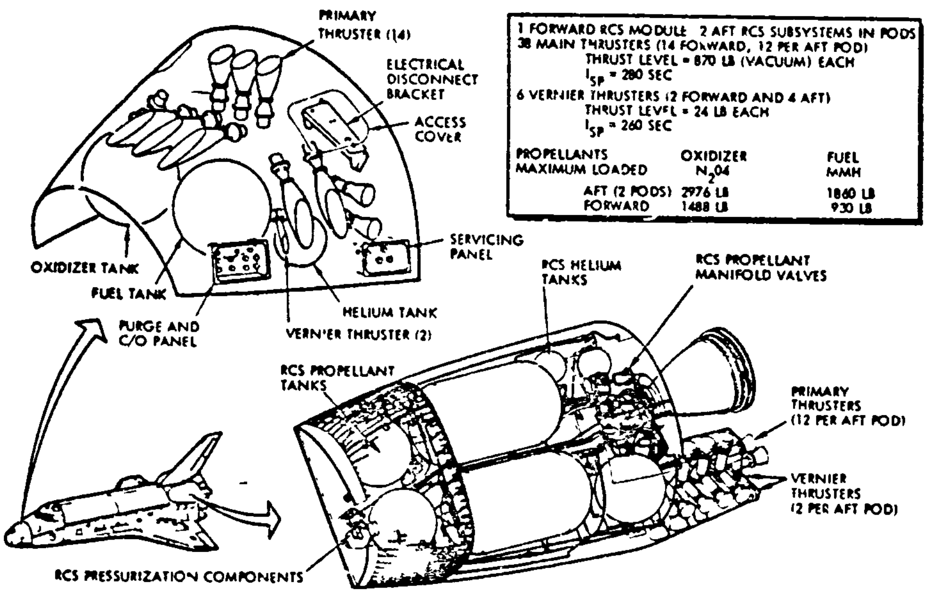

FIGURE 2.3 REACTION CONTROL SUBSYSTEM

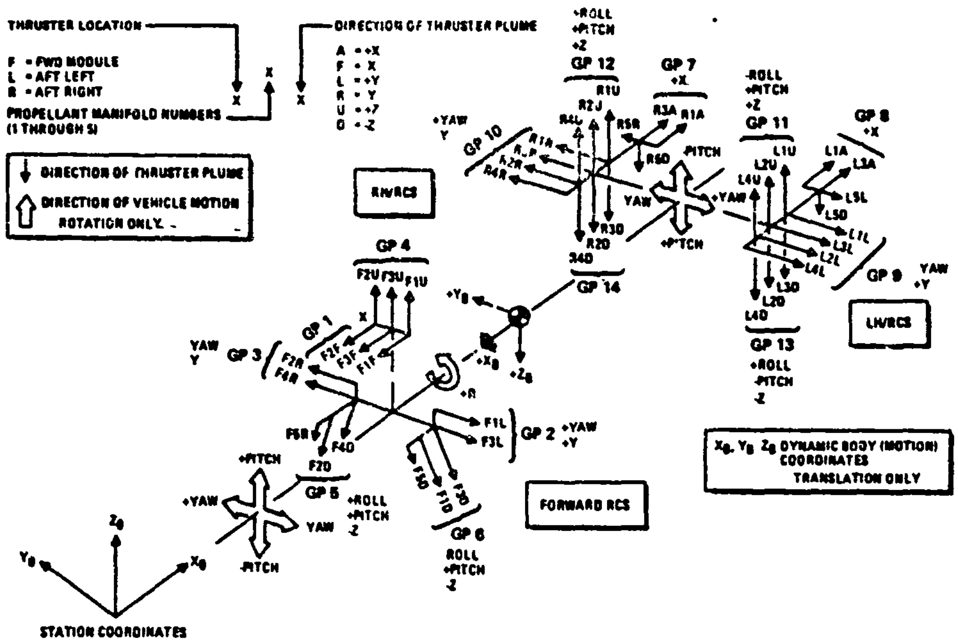

FIGURE 2.4 RCS THRUSTER IDENTIFICATION

TABLE 2.1 RCS THRUST COMPONENTS

THRUSTER NUMBER		THRUST COMPONENTS LB[a]			RESULTANT THRUST LB	THRUST APPLICATION[b]		
		FX_B	FY_B	FZ_B		X_O	Y_O	Z_O
F2F	NO 1	-879 4	-26 2	119 9	887 9	306.72	14 65	392 96
F3F		-879 5	0 0	122 7	888 0	306 72	0 0	394 45
F1F		-879 4	26.2	119 9	887 9	306 72	-14 65	392 96
F1L	NO 2	-26 3	873.6	18 2	874 2	362 67	-69 50	373 73
F3L		-21 0	870 3	0 5	870 6	364 71	-71 65	359 25
F2R	NO 3	-26.3	-873.6	18 2	874 2	362 67	69 50	373 73
F4R		-21 0	-870 3	0 5	870 6	364 71	71 65	359 25
F2U	NO 4	-32 3	-11 7	874 4	875 1	350 93	14 39	413 46
F3U		-31 9	0 0	873.5	874 1	350 92	0 0	414 53
F1U		-32 3	11 7	874 4	875 1	350 93	-14 39	413 40
F2D	NO 5 & NO 6	-28 0	-616 4	-639 5	898 6	333 84	61 42	356 95
F1D		-28 0	616 4	-639 5	898 6	333 84	-61 42	356 95
F4D		-24 8	-612 6	-639 4	885 9	348 44	66 23	358 44
F3D		-24 8	612 6	-639 4	885 9	348 44	-66 23	358 44
F5R		-0 8	-17 0	-17 6	24 5	324 35	59 70	350 12
F5L		-0 8	17 0	-17 6	24 5	324 35	-59 70	350 12

[a]MOTION COORDINATES
[b]STATION COORDINATES

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TABLE 2.1 RCS THRUST COMPONENTS (CONT.)

THRUSTER NUMBER		T' COMPONENTS, LB[a]			RESULTANT THRUST LB	THRUST APPLICATION[b]		
		FX_B	FY_B	FZ_B		X_O	Y_O	Z_O
R3A	NO 7	856 8	0.0	151 1	870 0	1555 29	137 00	473 00
R1A		856.8	0 0	151 1	870.0	1566.29	124 00	473 00
L3A	NO 8	856.8	0 0	151 1	870 0	1565 29	-137 00	473.06
L1A		856.8	0 0	151 1	870 0	1565 29	-124 00	473 06
L4L	NO. 9	0.0	870 5	-22 4	870 8	1516 00	-149 87	459 00
L2L		0 0	870 5	-22.4	870 8	1529 00	-149 87	459 00
L3L		0 0	870.5	-22 4	870 8	1542 00	-149 87	459 00
L1L		0.0	870.5	-22 4	870.8	1555 00	-149.87	459 00
R4R	NO. 10	0 0	-870 5	-22 4	870 8	1516 00	149 87	459 00
R2R		0 0	-870 5	-22 4	870 8	1529 00	149 87	459 00
R3R		0 0	-870.5	-22 4	870 8	1542 00	149 87	459 00
R1R		0 0	-870.5	-22.4	870 8	1556.00	149 87	459 00
L4U	NO 11	0 0	0 0	870 0	870 0	1516 00	-132 00	480 50
L2U		0 0	0 0	870 0	870 0	1529 00	-132.00	480 50
L1U		0.0	0 0	870 0	870 0	1542 00	-132 00	480 50
R4U	NO 12	0 0	0.0	870 0	870 0	1516.00	132 00	480 50
R2U		0 0	0 0	870 0	870 0	1529 00	132 00	480 50
R1U		0 0	0 0	870 0	870 0	1542 00	132 00	480 50
L4D	NO 13	170.4	291 8	-801 7	870.0	1516 00	-111 96	437 40
L2D		170.4	291 8	-801 7	870 0	1529 00	-111 00	440 00
L3D		170 4	291 8	-801 7	870.0	1542 00	-110 06	442 60
R4U	NO 14	170.4	-291 8	-801 7	870 0	1516.00	111 96	437 40
R2D		170 4	-291 8	-801 7	870 0	1529 00	111 00	440 00
R3D		170 4	-291 8	-801 7	870.0	1542 00	110 06	442 60
L5D		0 0	0 0	-24 0	24.0	1565.00	-118 00	455 44
R5D		0 0	0.0	-24 0	24.0	1565 00	118 00	455 44
L5L		0 0	24 0	-0 6	24 0	1565 00	-140 87	439 00
R5R		0 0	-24 0	-0 6	24 0	1565.00	149 87	459 00

[a]MOTION COORDINATES [b]STATION COORDINATES

Digital Autopilot (DAP)

The digital autopilot consists of a set of functional codes within the overall flight computer software. It provides the reconfiguration logic for switching between the various control modes, the RCS control laws including phase plane relationships/characteristics and jet select logic and the TVC control laws for OMS thrusting. The DAP in conjunction with its input/output electronics, computer hardware and the RCS jets provide three-axis stabilization as well as a number of automatic or manual flight modes for use in various on-orbit operations. The following paragraphs summarize these flight modes and their expected performance capabilities.

The main flight control modes are identified in the diagram of Figure 2.5 and are selected or engaged with the push button indicator (PBI) panel pictured in Figure 2.6. Mode select panels are located at the aft crew station as well as on the center console of the flight deck.

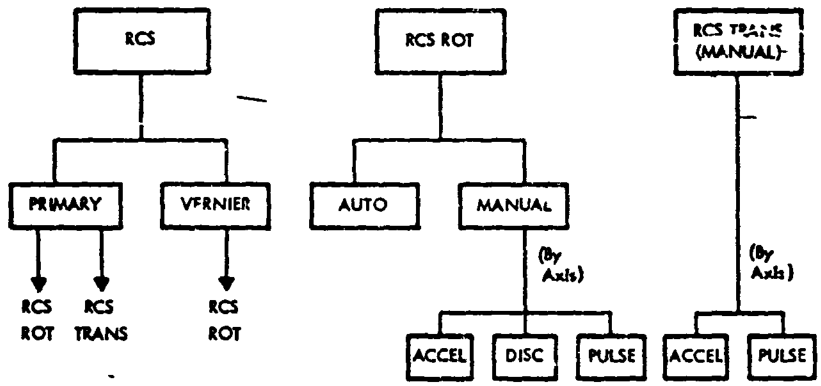

FIGURE 2.5 FCS CONTROL MODES

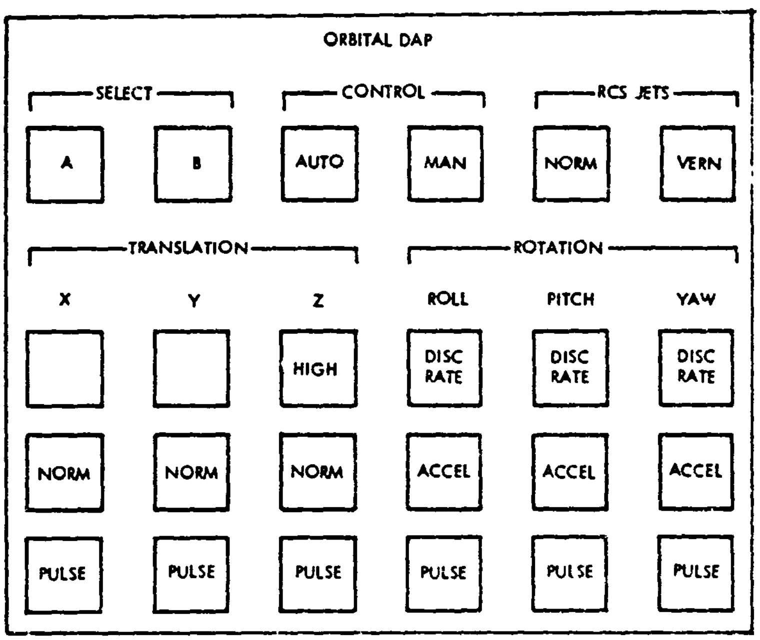

FIGURE 2.6 MANUAL FCS MODE SELECT (PBI) PANEL

Primary or Vernier Jet Selection. Primary or vernier jets are selected using the PBI panel. Pushing the "NORM" button selects the primary RCS and the "VERN" button selects the verniers. Selection of the vernier jets eliminates the possibility of performing translation maneuvers. The vernier system consists of only six jets and because of their installation geometry the system is not capable of producing "pure" translational motions. If translational motion is required as in the case for docking maneuvers, the primary thrusters must be selected. Further, it is not possible to simultaneously use the vernier jets for rotation and the primary jets for translation. If translation is required the primary thrusters must also be used for rotational control.

Automatic or Manual Rotation Control. Either of these modes is selected using the "AUTO" or "MAN" buttons on the PBI. The "AUTO" button engages the automatic mode and "MAN" the manual mode. These selections affect only the rotational control submodes. The translational control modes are always manual, even if the "AUTO" rotational mode is selected. In the automatic mode, the RCS (either primary or vernier) provides rotation control required to implement attitude and angular rate commands generated by the guidance system. Through the CRT display and keyboard it is also possible to command various desired orientations and then use the AUTO mode to implement the reorientation maneuver. Large maneuvers performed in this fashion utilize rotation about a single axis (eigenaxis), not necessarily one of the vehicle control axes. It is also possible, using the CRT/keyboard and the AUTO mode, to maneuver the orbiter to any specified orientation and then hold that attitude in either a local or inertial coordinate system. Thus, as an aid to docking maneuver operations the orbiter can be co-aligned with the SOC in a local vertical hold (LVH) coordinate frame, which then automatically maintains the SOC-orbiter alignment thereby freeing the pilot workload from rotational control tasks.

The manual RCS rotation mode is engaged if the manual PBI is pressed or the rotational hand controller (RHC) is out of detent in any axis. There are three manual rotation submodes as described below. Selection (but not activation) among these submodes can be enabled independently for each control axis through the nine PBIs on the mode select panel. If the flight control system is in the manual RCS rotation mode, the current PBI-selected submodes are active, and submode changes can only be made if the RHC is in detent in all axes. If the flight control system is not in the manual RCS rotation mode, the current PBI-selected submode will be indicated for each vehicle rotation axis by illumination of the appropriate rotational PBI. Thus, three rotation PBIs, one for each axis, will always be illuminated.

Manual Rotation - Acceleration/Drift Submode. This submode is independently selectable for each of the three vehicle control axes by pushing the "ACCEL" button on the PBI panel. In this submode, the RCS will implement a continuous angular acceleration about the affected axis, corresponding in direction to the sense of the RHC deflection, while the RHC is out of detent. Free drift will be implemented while the RHC is in detent.

Manual Rotation - Discrete Rate/Attitude Hold/Acceleration/Drift Submode. This submode is independently selectable for each of the three vehicle control axes by pushing the "DISC RATE" button on the PBI panel. In this submode, the type of rotational control will be a function of the position and history of the RHC. When the RHC is in the detent position, the RCS will implement an attitude hold about the affected axis. The reference attitude will be the attitude existing at the time the RHC was moved into detent or at the time the submode was initiated, whichever occured later. When the RHC is moved out of detent but not allowed to go beyond soft stop, the RCS will implement a preselected angular rate (discrete rate) about the affected axis, corresponding in direction to the sense of the RHC deflection. When the RHC is moved beyond soft stop, the RCS will implement continuous angular acceleration about the affected axis. When the RHC is moved back within soft stop but not allowed to go into detent, free drift about the affected axis will be implemented.

In essence, this submode provides the ability to manually change the orbiter attitude at preselected rates, but with the further capability of increasing the present maneuver rate by going beyond the soft step position. Attitude hold at the new attitude is then implemented by returning the RHC to the detent position.

Manual Rotation - Pulse/Acceleration/Drift Submode. This submode is independently selected for each of the three vehicle control axes by pushing the "PULSE" button on the PBI panel. In this submode, the type of rotational control will be a function of the position and history of the RHC. Each time the RHC is moved out of the detent but not allowed to go beyond the soft stop, the RCS will increment the angular rate about the affected axis by a preselected amount (pulse) corresponding in sign to the sense of the RHC deflection. It is possible to accumulate an integral number of desired angular rate increments by moving the RHC alternately out of detent and into detent while not allowing it to go beyond the soft stop. Free drift about the affected axis will occur following achievement of the accumulated desired angular rate increments, regardless of whether the RHC is in or out of detent at the time. When the RHC is moved beyond the soft stop, the RCS will implement continuous angular acceleration about the affected axis, corresponding in direction to the sense of RHC deflection. When the RHC is moved back within the soft stop, free drift about affected axis is implemented. The accumulated desired angular rate increments are zeroed whenever the RHC is moved beyond the soft stop so that free drift continues at the final accumulated rate resulting from the sequence of pulses plus the increment from the continuous angular acceleration interval. Stopping or slowdown actions are performed by reversing the RHC deflections.

This submode provides the ability to establish desired angular rates by combinations of continuous angular acceleration intervals and "beeping" in adjustments with pulses. Basic orbiter attitude changes could be made with this mode, but it is more useful in setting up desired attitude rates such as might be used in orbiter fly around operations for pre-docking inspections, etc.

In addition to the automatic and manual modes of rotational control described above there is additional versatility in the flight control system to allow the designation of nose or tail thrusters to perform pitch and/or yaw maneuvers. These options are called "LO NOSE" and "LO TAIL" and are entered with the CRT/keyboard panel. They may be utilized in conjunction with any of the rotational control modes employing the primary RCS system. The primary purpose of these options is for propellant management to assure balanced consumption between the forward and aft RCS modules. These modes would also provide smaller minimum impulses to rotational control operations, but would introduce larger translational coupling disturbances than the normal rotational control modes.

Manual Translation Control. As indicated earlier, RCS manual translation control is available only if the primary RCS is active (not if the verniers are selected). Once the primary RCS option is selected manual translation control is engaged, regardless of whether the manual or the automatic rotational control mode is selected. Thus, manual translational control is available at all times when the primary RCS is activated.

There are two manual translation submodes, as described below. Selection (but not activation) of either of these submodes may be enabled independently for each vehicle translation control axis, through a set of seven translation PBIs. Submode changes can only be made when the translation hand controller (THC) is in detent in all axes.

Manual Translation - Acceleration/Drift Submode. This submode is selected by pushing the "NORM" button on the PBI panel for the X and Y axes and by pushing either the "NORM" or "HIGH" buttons for the Z axis. In this submode, when the THC is moved out of the detent, the RCS will implement continuous acceleration along the affected axis, corresponding in direction to the sense of the THC deflection. When the THC is returned to the detent position, free drift occurs along the affected axis. If the "HIGH" option is selected for the Z axis, the +Z acceleration will be at a high level (up to 9 Z-thrusters firing) while the -Z acceleration will remain at the normal level (typically three thrusters firing). Thus for docking operations target closing maneuvers (-Z direction) are at normal acceleration levels while reverse thrusting (+Z direction) for abort purposes would be at the high acceleration levels.

Manual Translation - Pulse/Drift Submode. This submode is selected by pushing the "PULSE" button on the PBI panel. In this submode, the type of translational control is a function of the position and history of the THC. Each time the THC is moved out of the detent position, the RCS will increment the translational velocity along the affected axis by a preselected amount (pulse) corresponding in sign to the sense of the THC deflection. It is possible to accumulate an integral number of desired translational velocity increments by moving the THC alternately out of detent and into detent. Free drift along the affected axis occurs following achievement of the accumulated desired translational velocity increments, regardless of whether the THC is in or out of detent at the time.

Translation Control - "Low Z" Braking. In addition to the basic translation modes described above an additional option called "Low Z" braking (sometimes referred to as "±X braking) is also available. This auxiliary mode is selected with a special PBI switch and is intended for use in proximity operations where sensitivities to plume impingement are high. In this mode the plus and minus X RCS thrusters are fired simultaneously. The canted installation geometry of those thrusters provides a substantial +Z component of thrust (approximately 15 percent). By firing both forward and aft thrusters simultaneously their X-axis components cancel and the +Z components provide braking acceleration along the Z-axis. Since the main RCS plume concentrations are directed away from the region over the cargo bay, plume effects are reduced for both stationkeeping operations within the RMS reach envelope and docking approaches along the Z-axis.

Mode Compatibility. Figure 2.7 shows the compatible combinations of the on-orbit flight control modes. Basically, the main compatibilities are: (1) With the vernier RCS engaged, all rotational control modes and submodes can be activated, one at a time, with any mix of submodes per control axis that may be desired; (2) The same is true for the primary RCS selection, but with the added capability to mix any translational control mode/submode with any rotational mode/ submode configuration. The main incompatibilities are centered within the submodes where only one submode per axis is permitted in each of the rotational and translational control functions. This is because each submode is designed to produce a unique control action and simultaneously engagement would destroy the unique response. The other principal incompatibility is the primary/vernier jet selection. It is not possible to mix the use of these options such as to use the vernier jets for rotation control and the primary jets for translation control. Only rotational control is available with the verniers. If translation is needed, the primary RCS must be selected and rotational control is automatically switched to the primary RCS thrusters.

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REFERENCE PARAGRAPH†
(3.3 1 3--)

REF PAR		2.1	2.2	2.3	2.3.1	2.3.2	2.3.2.1	2.3.2.2	2.3.2.3	2.4	2.4.1	2.4.2
2.1	PRIMARY RCS	I	X									
2.2	VERNIER RCS	X	I							X	X	X
2 3	RCS ROTATION			I								
2.3.1	AUTO RCS ROTATION (EIGENAXIS)				I	X	X	X	X			
2 3.2	MANUAL RCS ROTATION				X	I						
2.3.2 1	ACCELERATION/DRIFT				X		I	*	*			
2.3 2 2	DISC/A.H /ACCEL/DRIFT				X		*	I	*			
2.3.2.3	PULSE/ACCEL/DRIFT				X		*	*	I			
2.4	MANUAL RCS TRANSLATION		X							I		
2 4 1	ACCELERATION/DRIFT		X								I	*
2 4 2	PULSE/DRIFT		X								*	I

†PARAGRAPH NUMBERS IN THE REFERENCE DOCUMENT· SD 72-SH-0105, VOLUME I, BOOK 2, *REQUIREMENTS DEFINITION DOCUMENT* (FEB 1980).

KEY

BLANK = COMPATIBLE

* = COMPATIBLE WITHIN A GIVEN AXIS

X = INCOMPATIBLE

I = IDENTITY

FIGURE 2.7 FCS MODE COMPATIBILITY

Mode Set A or B. Within the overall framework of modes and submodes discussed above there exists an additional option for rapidly changing the on-orbit control configuration. The PBI panel (previous Figure 2.6) has an A or B selection option in the upper left hand corner. This allows rapid switching between two preselected control configurations. These control configurations consist of specific combinations of rotational and translational modes each with specified attitude deadbands, angular rates, angular or translational pulse sizes, etc. This could be used, for example, to assist in making changes from inertial attitude holds to local vertical holds or in reducing attitude deadbands and translation pulse sizes during the final phases of terminal closure to aid in precision trajectory control. Thus, it can be used in many ways and circumstances to reduce pilot workload during critical mission operations.

On-Orbit Control Performance. As indicated in the preceding paragraph the overall control configuration can be rapidly changed during the mission with the A/B mode set selection buttons on the PBI panel. In addition, most of the control parameters in these pre-specified mode sets can also be individually changed during the mission with the CRT/keyboard panel. The selection ranges for those parameters are listed in Table 2.2. Although the

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TABLE 2.2 DAP LOAD PARAMETERS

PARAMETER	UNIT	EACH DAPLOAD CONTAINS SEPARATE VALUES FOR		VALUES AUTOMATICALLY UPDATED WHEN VEH INERTIA DIAGONAL UPDATED	MANUALLY SELECTABLE RANGE OF KEYBOARD ENTRIES
		PRIM/VERN RCS	ROLL/PITCH YAW		
• THRESHOLD FOR OPEN-LOOP RCS OFF-AXIS COUPLING COMPENSATION FIRING	DEG/SEC	YES	NO	NO	PRIM: 0 00 TO 0 99 VERN: 0 00 TO 0 999
• ATTITUDE DEADBAND, RCS	DEGREE	NO	YES	NO	0 010 TO 40 000
• MANEUVER RATE, AUTOMATIC AND MANUAL DISCRETE RATE RCS ROTATION	DEG/SEC	YES	NO	NO	PRIM: 0 050 TO 2 000 VERN: 0 002 TO 1 000
• PITCH ACCELERATION OPTION (NOMINAL, LOW USING FWD RCS, LOW USING AFT RCS)	NONE	NO	NO	NO	1 TO 3
• YAW ACCELERATION OPTION (NOMINAL, LOW USING FWD RCS, LOW USING AFT RCS)	NONE	NO	NO	NO	1 TO 3
• PAYLOAD CONFIGURATION (PAYLOAD NOT EXTENDED ANY OF FIVE PAYLOAD-EXTENDED CONFIGURATIONS)	NONE	NO	NO	NO	0 TO 5
• RATE DEADBAND, RCS ROTATION	DEG/SEC	YES	NO	NO	PRIM 0 2 TO 5 00 VERN: 0 01 TO 0 500
• MANUAL RCS ROTATION PULSE SIZE	DEG/SEC	YES	NO	NO	PRIM: 0 04 TO 1 00 VERN: 0 .01 TO 0 500
• MANUAL RCS TRANSLATION PULSE SIZE	FT/SEC	NO	NO	NO	0 01 TO 5 00
• NOMINAL AVERAGE CONTROL ACCELERATION RCS ROTATION, PAYLOAD NOT EXTENDED, PRIMARY RCS	DEG/SEC^2	NO	YES	YES	TO BE DETERMINED (TBD)
• NOMINAL AVERAGE CONTROL ACCELERATION, RCS ROTATION, PAYLOAD NOT EXTENDED VERNIER RCS	DEG/SEC^2	NO	YES	YES	TBD

DAP load parameter ranges listed in this table can be entered via the keyboard the actual system performance may be degraded slightly by various hardware characteristics such as minimum impulse sizes, motion coupling, sensor tolerances and alignments and other factors related to the many individual elements which make up the flight control system. The current expected performance capabilities are summarized in the following chart segments.

o ANGULAR RATE LIMITS:

RCS JETS	RATE LIMIT RANGE (°/SEC)
PRIMARY	0.2 to 4.0
VERNIER	0.01 to 0.5

o ATTITUDE DEADBANDS:

RCS JETS	ATTITUDE DEADBAND RANGE (DEGREES)
PRIMARY	1.0 to 20.0
VERNIER	0.1 to 20.0

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o LIMIT CYCLE RATES:

RCS JETS	MAX. LIMIT CYCLE RATE (°/SEC)
PRIMARY	0.2
VERNIER	0.01

o DISCRETE ANGULAR RATES:

RCS JETS	DISCRETE RATE RANGE (°/SEC)	TOLERANCE
PRIMARY	0.1 to 2.0	± RATE LIMIT
VERNIER	0.1 to 1.0	± RATE LIMIT

o TRANSLAION PULSE SIZE:

RCS JETS	PULSE SIZE RANGE	TOLERANCE
PRIMARY	1.5 to 150.0 CM/SEC (0.05 to 5.0 FT/SEC)	± 1.5 CM/SEC (± 0.05 FT/SEC)
VERNIER	NO TRANSLATION CONTROL	

These performance capabilities are the current "work to" numbers for STS-1 and 2 (Reference 7). Actual flight experience and further development efforts will likely yield improvements as the STS system matures. Although this improved performance will likely be available for SOC related missions, the actual performance which will be achieved is unknown. Thus, for conservatism, the current performance characteristics listed here were used in this study.

2.2.3 Simplified "Open-Loop" Docking Analysis

The preceding paragraphs have identified the many factors which can affect the ability of the orbiter to dock with the SOC. A simplified "open loop" simulation analysis of the terminal docking maneuver was conducted to develop a general feel for the relative significance and/or sensitivity to some of these influencing factors. Specifically, the effects of approach path direction, various flight control system characteristics, orbiter attitude, c.g. offset of the docking port and the interactions of these factors with orbit mechanics effects were investigated.

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The basic problem model for this analysis is illustrated in Figure 2.8. Using these input characteristics, a series of ballistic closing paths were computed for initial standoff distances of 9.1M (30ft) and 15.2M (50ft) on the High Fidelity Relative Motion Program (HFRMP). This simulation program models the relative motion of the orbiter and a coorbiting vehicle (SOC) and outputs orbital and relative motion parameters. The location and force components of each of the 44 primary and vernier RCS thrusters are contained in the program. They are used to model the orbiter translational and

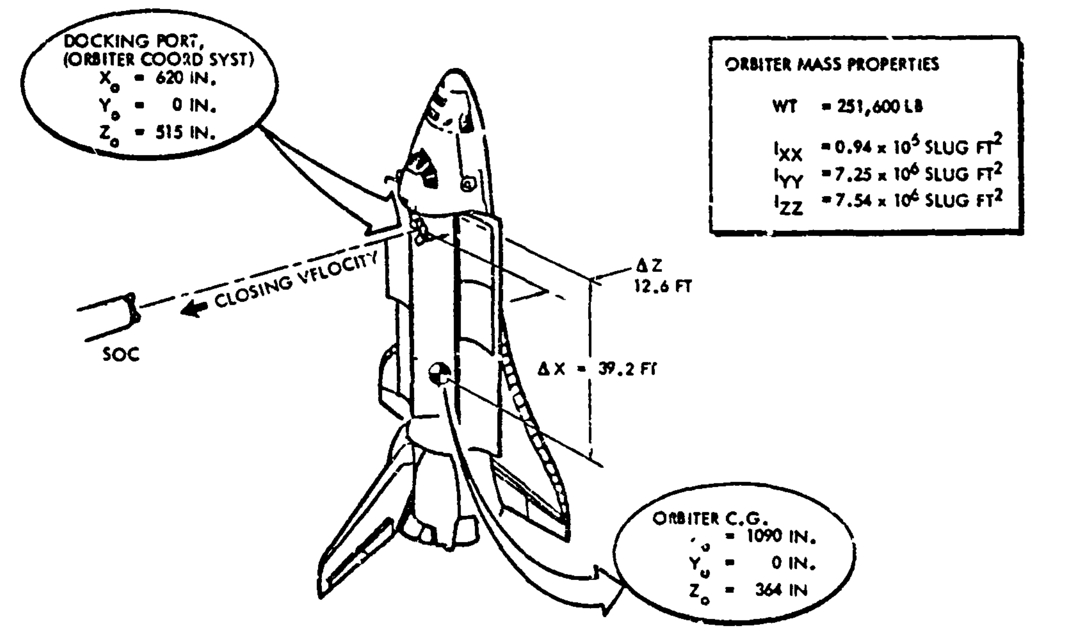

FIGURE 2.8 DOCKING TRAJECTORY GEOMETRY

rotational maneuvers currently possible, along with the propellant consumption. The program also models the aero and gravity gradient effects as influenced by orbiter attitude and the orbit mechanics interactions induced by c.g. offsets and approach path variations.

An idealized closing ΔV was determined for each simulation case to be run. These ΔVs were designed to carry the orbiter from its initial standoff location to a "perfect" docking contact closure. The assumption was made that initially the orbiter docking port was aligned with the SOC docking port along the approach path (V-bar or R-bar) with zero relative motion between the docking ports. This implies the capability for accurate orbiter stationkeeping at the start of the problem even though the c.g. offset geometry places the two vehicles in slightly different orbits. Ideal closing times of 1.8 and 3.0 minutes were selected for the 9.1M to 15.2M standoff conditions respectively. This gives nominal contact velocities in the upper mid range of the docking envelope specified for SOC in Figure 2.9.

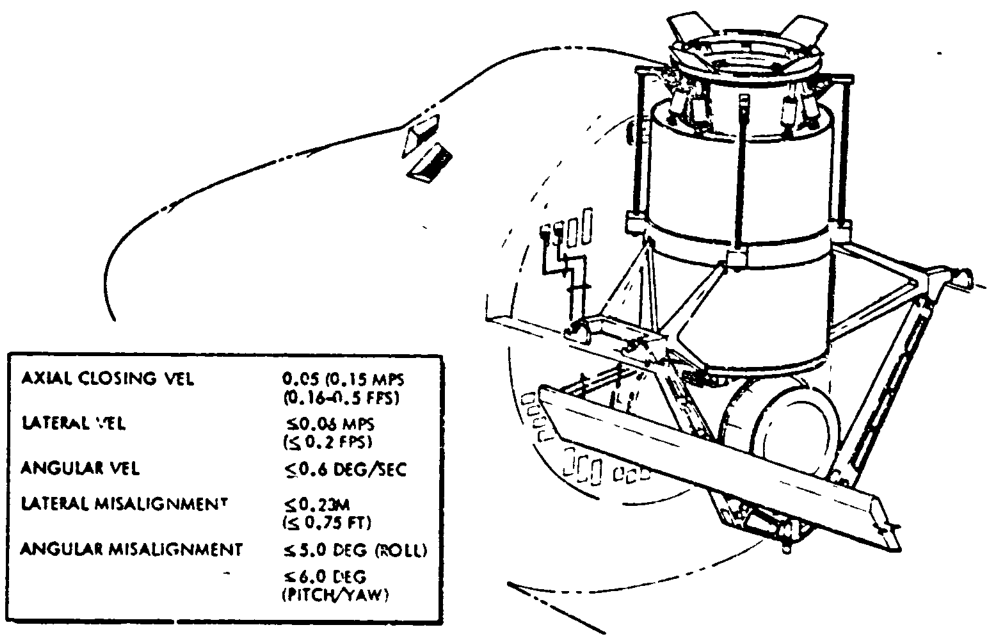

FIGURE 2.9 PRELIMINARY SOC DOCKING REQUIREMENTS

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However, the effects of slight imperfections on the closing maneuvers were superimposed on these ideal conditions. First, the actual thruster burn times for the X and Z thrusters were determined allowing for their canted thrust angles and c.g. offsets. In some cases one or two corrective rotational impulses were also introduced to simulate coupling compensation effects. Various combinations of plus or minus 80 milliseconds of thrusting were superimposed on these idealized burn times. These would be the worst case errors which can be introduced by the 80 millisecond minimum impulse size mechanized in the DAP. Figure 2.10 shows an example CRT output graphic from the HFRMP program for one of the cases. The coordinate system is shown

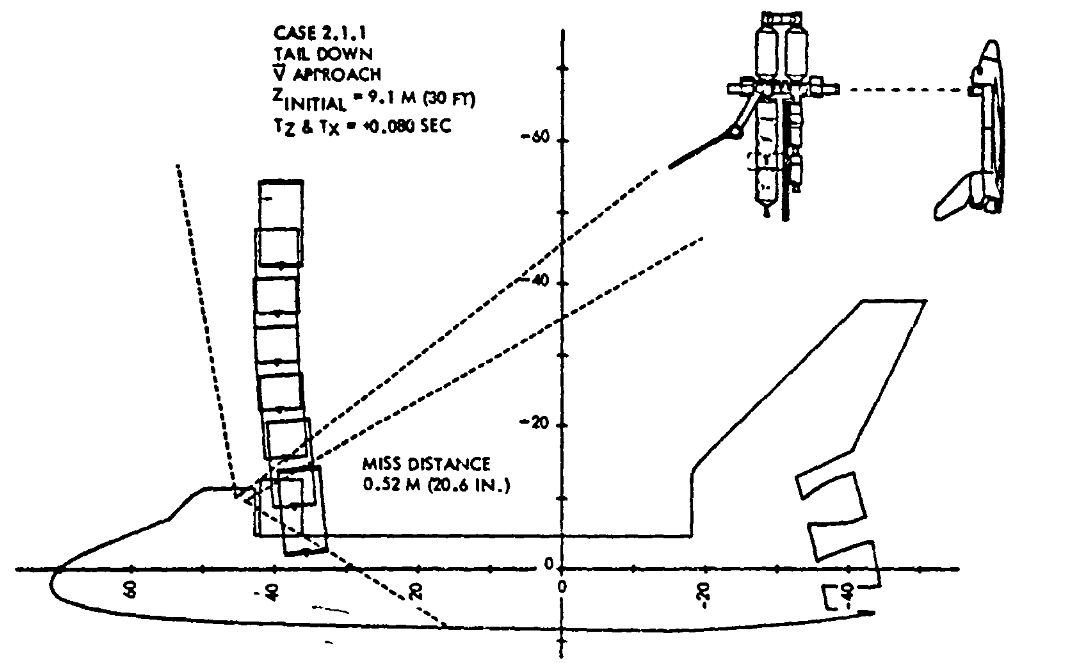

FIGURE 2.10 EXAMPLE DOCKING TRAJECTORY

at the orbiter c.g. and the sequence of relative positions between the orbiter and the SOC are symbolized by the series of overlapping rectangles arcing toward the orbiter docking port. The rectangles represent a section of the SOC service module. As noted on the figure, the case shown is for a tail down approach along the velocity vector (V-bar) with an initial standoff distance of 9.1M and with X and Z burn time errors of plus 80 milliseconds The docking miss distance is shown to be 0.52M (20.6 in), significantly outside the 0.23M (±9 in) envelope allowed (Figure 2.9).

The results for all of the cases run are summarized in Tables 2.3 through 2.6. The idealized velocities are noted on these tables along with the individual firing time error combinations, fuel used, actual post burn ΔVs and the docking contact conditions. The docking contact conditions are measured at the docking port and include the effects of angular motions coupled with the c.g. offsets as well as the basic orbiter c.g. motion. The numbering code for the individual case runs in the left hand column of these charts is described in Table 2.7 below. The maximum orbiter attitude deviations and deadband characteristics for each run are shown in Tables 2.8 and 2.9 respectively. CRT graphics similar to the example in Figure 2.10 are contained in Appendix B for all the cases run. Appendix B presents the complete simulation analysis package.

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TABLE 2.3 30-FT, $+\bar{V}$ NOSE DOWN, 1.8 MINUTE TERMINAL APPROACH CASE*

CASE	FIRING TIME (sec)		FUEL USED (lb)	POST BURN V's (ft/sec)		DOCKING PARAMETERS WRT SS BODY AXES CENTERED ON SS DOCK							
	$-X_b$	$-Z_b$		$-V_{xb}$	V_{zb}	T_{dock} (sec)	V_z (ft/s)	$\sqrt{V_x^2 + V_y^2}$ (ft/sec)	$\sqrt{\omega_x^2 + \omega_y^2 + \omega_z^2}$ (deg/sec)	$\sqrt{x^2 + y^2}$ (inches)	$\lvert\theta_x\rvert$ (deg)	$\lvert\theta_y\rvert$ (deg)	$\lvert\theta_z\rvert$ (deg)
1 1 1	T_1+0 08	T_1+0 08	18 1	0 006	0 329	91 0	0 330	0 036	0 035	16 7	0 49	3 08	0 02
1 1 2	T_1+0.08	T_1+0 08	18 8	0 006	0 389	96 0	0 239	0 033	0 027	11 8	0 53	1 91	0 01
1 1 3	T_1+0 08	T_1+0 08	17.7	0 073	0 272	111 0	0 260	0 082	0 054	7 6	0 67	6 06	0 02
1 2 1	T_1-0 08	T_1+0 08	17 1	0 031	0 344	88 7	0 348	0 005	0 049	19 3	0 48	4 27	0 02
1 2 2	T_1-0 08	T_1+0 08	17 8	0 031	0 344	97 1	0 312	0 004	0 013	26 0	0 54	2 43	0 01
1 2 3	T_1-0 08	T_1+0 08	16 08	0 099	0 287	102 8	0 293	0 035	0 040	38 9	0 62	4 14	0 02
►1 3 1	T_1-0 08	T_1-0 08	14 1	0 010	0 275	108 2	0 229	0 011	0 055	1 4	0 45	5 94	0 02
1 3 2	T_1-0 08	T_1-0 08	14 8	0 010	0 275	117 0	0 241	0 011	0 007	9 9	0 52	2 98	0 01
1 3 3	T_1-0 08	T_1-0 08	13 1	0 078	0 218	135 5	0 217	0 055	0 033	18 0	0 67	4 54	0 03
1 4 1	T_1+0 08	T_1-0 08	15 1	0 015	0 260	115 [illegible]	0 261	0 053	0 041	46 8	0 49	4 79	0 02
1 4 2	T_1+0 08	T_1-0 08	14 1	0 015	0 260	126 4	0 214	0 057	0 020	41 6	0 57	1 20	0 00
1 4 3	T_1+0 08	T_1-0 08	14 1	0 52	0 203	153 7	0 175	0 107	0 046	4 4	0 75	7 30	0 02
►1 5 1	T_1+0 08	T_1+0 08	17 8	0 018	0 327	90 8	0 330	0 025	0 030	4 2	0 39	2 66	0 02
►1 5 2	T_1+0 08	T_1+0 08	18 6	0 18	0 327	96 4	0 289	0 025	0 031	1 3	0 43	0 73	0 01
1 5 3	T_1+0 08	T_1+0 08	16 8	0 090	0 270	112 0	0 264	0 079	0 058	6 9	0 56	6 62	0 02
1 6 1	T_1+0 08	T_1-0 08	14 8	0 040	0 260	116 9	0 260	0 048	0 038	31 8	0 39	4 40	0 02
1 6 2	T_1+0 08	T_1-0 08	15 6	0 040	0 260	126 5	0 216	0 049	0 022	27 3	0 45	0 81	0 01
1 6 3	T_1+0 08	T_1-0 08	13 9	0 060	0 200	152 2	0 180	0 102	0 049	36 4	0 61	7 68	0 02
►1 7 1	T_1-0 08	T_1-0 08	14 3	0 010	0 270	112 4	0 270	0 026	0 045	7 3	0 33	5 04	0 02
►1 7 2	T_1-0 08	T_1-0 08	15 1	0 010	0 270	120 9	0 230	0 026	0 016	1 5	0 43	0 91	0 02
1 7 3	T_1-0 08	T_1-0 08	13 4	0 000	0 210	142 0	0 200	0 072	0 042	8 5	0 57	6 16	0 02
1 8 1	T_1-0 08	T_1+0 08	17 3	0 030	0 330	82 2	0 340	0 001	0 037	9 6	0 35	3 00	0 02
1 8 2	T_1-0 08	T_1+0 08	18 1	0 030	0 330	99 4	0 300	0 029	0 024	19 8	0 45	1 32	0 01
1 8 3	T_1-0 08	T_1+0 08	16 3	0 083	0 291	10[illegible]	0 [illegible]	0 051	0 051	29 1	0 53	5 49	0 02

*IDEAL ΔV's REQUIRED $V_{\bar{Z}}$ = 0 3430 ft/sec $-V_{\bar{R}}$ = 0 0422, TWO NOSE PRIMARY THRUSTERS USED FOR X_b TRANSLATION

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TABLE 2.4 30-FT, +$\bar{V}$ NOSE UP, 1.8 MINUTE TERMINAL APPROACH CASE*

CASE	FIRING TIME (sec)		FUEL USED (lb)	ΔV (ft/sec)		DOCKING PARAMETERS WRT SS BODY AXES CENTERED ON SS DOCK							
	t_b	$-t_b$		V_{xb}	V_{zb}	T_{DOCK} (sec)	$\dot{z}$ (ft/s)	$\sqrt{\dot{x}^2+\dot{y}^2}$ (ft/sec)	$\sqrt{\omega^2+\omega^2}$ (deg/sec)	$\sqrt{x^2+y^2}$ (inches)	$\lvert\theta_x\rvert$ (deg)	$\lvert\theta_y\rvert$ (deg)	$\lvert\theta_z\rvert$ (deg)
2 1 1	T_1+0 08	T_1+0 08	14 0	0 058	0 327	90 5	0 329	0 020	0 030	20 6	0 31	2 6	0 01
2 1 2	T_1+0 00	T_1+0 08	14 7	0 058	0 327	95 7	0 289	0 030	0 030	12 7	0 34	0 72	0 00
2 1 3	T_1+0 08	T_1+0 08	13 0	0 009	0 270	111 0	0 274	0 003	0 058	7 0	0 46	7 07	0 01
2 2 1	T_1-0 08	T_1+0 08	13 0	0 033	0 342	88 6	0 331	0 059	0 045	14 3	0 30	3 88	0 01
2 2 2	T_1-0 08	T_1+0 08	13 7	0 033	0 342	92 4	0 350	0 0/0	0 017	23 1	0 33	2 20	0 01
2 2 3	T_1-0 08	T_1+0 08	12 0	0 034	0 285	105 5	0 285	0 044	0 044	48 4	0 43	4 73	0 02
2 3 1	T_1-0 08	T_1-0 08	10 0	0 013	0 216	109 7	0 268	0 037	0 051	10 7	0 74	5 55	0 02
►2 3 2	T_1-0 08	T_1-0 08	10 7	0 054	0 273	119 0	0 230	0 046	0 010	4 8	0 20	2 51	0 01
2 3 3	T_1-0 08	T_1-0 08	9 0	0 054	0 273	138 4	0 217	0 004	0 036	31 0	0 41	5 25	0 02
2 4 1	T_1+0 08	T_1-0 08	11 0	0 080	0 258	113 6	0 368	0 004	0 037	54 6	0 27	4 38	0 01
2 4 2	T_1+0 08	T_1-0 08	11 7	0 012	0 201	119 0	0 228	0 056	0 010	19 4	0 23	2 50	0 01
2 4 3	T_1+0 08	T_1 0 08	10 0	0 012	0 201	147 1	0 207	0 016	0 050	25 7	0 44	7 66	0 01
2 5 1	T_1+0 08	T_1+0 08	13 7	0 048	0 327	91 6	0 326	0 029	0 028	10 2	0 24	2 49	0 01
►2 5 2	T_1+0 08	T_1+0 08	14 5	0 019	0 270	96 5	0 287	0 039	0 039	1 8	0 28	0 50	0 01
2 5 3	T_1+0 08	T_1+0 08	12 7	0 019	0 270	111 4	0 273	0 011	0 060	19 3	0 38	6 78	0 01
2 6 1	T_1-0 08	T_1+0 08	13 2	0 035	0 334	91 0	0 327	0 04[illegible]	0 035	7 7	0 24	2 19	0 01
2 6 2	T_1-0 08	T_1+0 08	14 0	0 035	0 334	94 8	0 291	0 039	0 025	16 5	0 26	1 24	0 01
2 6 3	T_1-0 08	T_1+0 08	7 0	0 032	0 277	108 3	0 280	0 032	0 053	40 4	0 32	5 82	0 01
2 7 1	T_1-0 08	T_1-0 08	10 3	0 058	0 266	112 5	0 266	0 028	0 043	19 8	0 19	4 79	0 01
►2 7 2	T_1-0 08	T_1-0 08	11 0	0 058	0 266	121 5	0 225	0 036	0 017	4 9	0 23	1 55	0 00
2 7 3	T_1-0 08	T_1-0 08	9 3	0 098	0 209	142 3	0 213	0 014	0 044	19 2	0 35	6 46	0 01
2 8 1	T_1+0 08	T_1-0 08	10 8	0 070	0 259	114 3	0 266	0 006	0 036	42 1	0 19	3 98	0 01
2 8 2	T_1+0 08	T_1-0 08	11 5	0 070	0 259	124 2	0 223	0 016	0 024	28 6	0 25	0 61	0 00
2 8 3	T_1+0 08	T_1-0 08	10 8	0 003	0 202	147 0	0 207	0 007	0 051	9 6	0 35	7 72	0 01

*IDEAL ΔV's REQUIRED $\Delta V_{\bar{V}}$ = 0 2098 ft/sec, $\Delta V_{\bar{R}}$ = -0 0258, TWO NOSE PRIMARY THRUSTERS USED FOR $-X_b$ TRANSLATION

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TABLE 2.5 + $\bar{R}$ 30-FT (1.8 MIN.) AND 50-FT (3 MIN.) TERMINAL APPROACH CASE*

	FIRING TIME (sec)			POST BURN V's (ft/sec)		DOCKING PARAMETERS WRT SS BODY AXES CENTERED ON SS DOCK							
CASE	X_b	$-Z_b$	FUEL USED (lb)	V_{xb}	$-V_{zb}$	T_{dock} (sec)	$\dot{r}$ (ft/s)	$\sqrt{V_x^2 + V_y^2}$ (ft/sec)	$\sqrt{\omega_x^2 + \omega_y^2}$ (deg/sec)	$\sqrt{x^2 + y^2}$ (inches)	$\lvert\theta_x\rvert$ (deg)	$\lvert\theta_y\rvert$ (deg)	$\lvert\theta_z\rvert$ (deg)
►3 5 4 1	T₁+0 08	T₁+0 08	15 1	0 002	0 285	100 0	0 331	0 018	0 039	3 7	0 42	1 49	0 04
3 6 4 1	T₁-0 08	T₁+0 08	14 5	0 014	0 292	98 2	0 333	0 036	0 046	22 6	0 38	0 85	0.04
3 7 4 1	T₁+0 08	T₁-0 08	12 1	0 021	0 217	120 0	0 266	0 005	0 047	28 7	0 35	0 40	0 04
►3 8 4 1	T₁-0 08	T₁-0 08	11 6	0 008	0 224	120 0	0 267	0 025	0 065	4 1	0 34	1 26	0 04
3 5 4 2	T₁+0 08	T₁+0 08	15 0	0 008	0 294	163 3	0 327	0 040	0 041	12 3	0 69	1 25	0 08
3 6 4 2	T₁-0 08	T₁+0 08	14 5	0 002	0 301	161 0	0 325	0 061	0 049	42 0	0 62	0 14	0 09
3 7 4 2	T₁+0 08	T₁-0 08	12 0	0 033	0 226	182 5	0 262	0 031	0 049	43 6	0 58	2 04	0 09
►3 8 4 2	T₁-0 08	T₁-0 08	11 5	0 023	0 233	180 0	0 258	0 052	0 058	5 2	0 57	3 44	0 10
►4 5 4 1	T₁+0 08	T₁+0 08	16 8	0 006	0 315	92 0	0 363	0 010	0 031	5 3	0 39	2 63	0 03
4 6 4 1	T₁-0 08	T₁+0 08	16 3	0 019	0 322	90 2	0 366	0 031	0 039	22 9	0 40	1 97	0 03
4 7 4 1	T₁+0 08	T₁-0 08	13 8	0 016	0 247	112 3	0 297	0 101	0 041	21 7	0 36	1 01	0 03
►4 8,4 1	T₁-0 08	T₁-0 08	13 3	0 003	0 255	110 1	0 299	0 021	0 047	1 3	0 37	0 50	0 04
►4 5 4 2	T₁+0 08	T₁+0 08	15 0	0 007	0 294	163 2	0 326	0 043	0 042	5 8	0 63	1 11	0 08
4 6 4 2	T₁-0 08	T₁+0 08	14 5	0 003	0 302	161 0	0 324	0 064	0 050	36 1	0 62	0 07	0 09
4 7 4 2	T₁+0 08	T₁-0 08	12 0	0 032	0 226	202 5	0 262	0 034	0 050	48 3	0 58	2 30	0 09
4 8 4 2	T₁-0 08	T₁-0 08	11 5	0 022	0 234	200 0	0 257	0 054	0 058	9 6	0 57	3 62	0 10

*1 NOSE PRIMARY THRUSTER USED FOR $-X_b$ TRANSLATION

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TABLE 2.6 X-POP (SIDEWAYS), +$\overline{V}$, 1.8 MIN., 30-FT APPROACH CASES*

CASE	FIRING TIME (sec)			FUEL USED (lb)	POST BURN ΔV's (ft/sec)			DOCKING PARAMETERS WRT SS BODY AXES CENTERED ON SS DOCK							
	$-X_b$	$-Y_b$	$-Z_b$		ΔV_{xb}	ΔV_{yb}	ΔV_{zb}	t_{dock} (sec)	$\dot{z}$ (ft/s)	$\sqrt{V_x^2+V_y^2}$ (ft/sec)	$\sqrt{\dot{\theta}_x^2+\dot{\theta}_y^2+\dot{\theta}_z^2}$ (deg/sec)	$\sqrt{x^2+y^2}$ (inches)	$\lvert\theta_x\rvert$ (deg)	$\lvert\theta_y\rvert$ (deg)	$\lvert\theta_z\rvert$ (deg)
<u>CASE (1)</u> SIDEWAYS WITH 0.08 SEC INITIAL ROLL, PITCH, AND YAW ROTATIONAL CORRECTIONS	T_i + 0.08	T_i + 0.08	T_i + 0.08	19.3	0.033	0.035	0.302	91.0	0.330	0.045	0.0?4	33.5	5.01	1.60	2.76
<u>CASE (2)</u> SIDEWAYS WITH ONE MIDCOURSE PITCH CORRECTION	T_i + 0.08	T_i + 0.08	T_i - 0.08	16.1	0.031	0.093	0.275	115.0	0.238	0.044	0.060	37.9	7.30	2.70	2.45
<u>CASE (3)</u> SAME AS (1)	T_i - 0.08	T_i - 0.08	T_i - 0.08	14.3	0.031	0.021	0.276	110.0	0.268	0.073	0.064	57.5	2.50	6.05	7.45
<u>CASE (4)</u> SAME AS (2)	T_i - 0.08	T_i - 0.08	T_i 0.08	15.1	0.031	0.021	0.276	118.0	0.230	0.075	0.046	14.1	1.42	0.97	1.25

*2 NOSE PRIMARY THRUSTERS USED FOR $-X_b$ TRANSLATIONS

TABLE 2.7 EXPLANATION OF CASE NUMBERINGS SYSTEM

FIRST NUMBER—APPROACH PATH AND ATTITUDE

1. $+\bar{V}$, nose down
2. $+\bar{V}$, nose up
3. $-\bar{R}$, nose (X_b) in $+\bar{V}$ direction
4. $+\bar{R}$, nose (X_b) in $-\bar{V}$ direction

SECOND NUMBER—FIRING TIME ERRORS

For a given case, numbers 1 through [illegible] designate the four [illegible] ways to fire the $-X_b$ and $-Z_b$ [illegible] thrusters, either [illegible] or the ideal firing time.

Additionally, using [illegible] number designates the same thing, [illegible]

THIRD NUMBER—A[illegible]

1. X_b and Z_b [illegible] burns [illegible] immediately followed by a 0.35-sec pitch rotational correction
2. Same as 1, with another 0.[illegible]-sec pitch rotational correction approximately halfway to the SOC de[illegible]
3. Same as 1, except no rot[illegible] correction[illegible]
4. Same as 1, except the pitch correction occurs [illegible]

FOURTH NUMBER

[illegible] 2 or no [illegible] closure

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TABLE 2.8 MAXIMUM ORBITER ATTITUDE ERRORS (WITH RESPECT TO THE IDEAL LOCAL VERTICAL ORIENTATION)

CASE NUMBER	MAXIMUM ERROR			
	RATE (°/sec)		EXCURSION (degrees)	
	PITCH	ROLL	PITCH	ROLL
1 1 1	056	006	3 08	49
1 1 2	056	006	2 03	53
1 1 3	056	006	6 06	67
1.2 1	049	006	4 27	48
1.2.2	049	006	2 86	54
1.2 3	041	006	4 14	62
1 3 1	055	005	5 94	45
1 3 2	.055	005	3 25	52
1 3 3	035	005	4 54	67
1 4.1	049	005	4 79	49
1 4 2	049	005	2 44	57
1.4 3	049	005	7 30	75
1 5.1	060	005	2 66	39
1 5.2	060	005	1 77	43
1 5 3	060	005	6 62	56
1 6 1	052	004	4 40	39
1 6 2	052	004	2 27	45
1 6 3	052	004	7 68	61
1 7 1	045	004	3 08	53
1 7 2	045	004	2 71	43
1 7 3	045	004	6 16	57
1 8 1	053	005	3 00	35
1.8 2	053	005	2 21	45
1 8 3	053	005	5 49	53

CASE NUMBER	MAXIMUM ERROR			
	RATE (°/sec)		EXCURSION (degrees)	
	PITCH	ROLL	PITCH	ROLL
2 1 1	060	004	2 66	31
2 1 2	060	004	1 77	34
2 1.3	060	004	7 07	46
2 2 1	046	004	3 88	30
2 2 2	046	004	2 65	33
2 2 3	046	004	4 73	43
2 3 1	051	003	5 55	24
2 3 2	051	003	3 05	20
2 3 3	[illegible]	003	5 25	41
2 4 1	054	003	4 38	27
2 4 2	054	003	2 50	23
2 4 3	054	003	7 66	44
2 5 1	062	003	2 49	24
2 5 2	062	003	1 64	28
2 5 3	062	003	6 78	38
2 6 1	055	003	3 19	24
2 6 2	055	003	2 09	25
2 6 3	055	003	5 82	32
2 7 1	047	002	4 79	19
2 7 2	047	002	2 58	23
2 7 3	047	002	6 46	35
2 8 1	054	002	3 98	19
2 8 2	054	002	2 14	25
2 8 3	054	002	7 72	35

CASE NUMBER	MAXIMUM ERROR			
	RATE (°/sec)		EXCURSION (degrees)	
	PITCH	ROLL	PITCH	ROLL
3 1 4 1	050	004	3 02	42
3 2 4 1	042	004	2 57	38
3 3 4 1	042	003	2 52	35
3 4 4 1	035	003	2 08	34
3 1 4 2	047	004	4 24	69
3 2 4 2	050	004	3 57	62
3 3 4 2	051	003	3 50	58
3 4 4 2	058	003	2 83	57
4 1 4 1	058	005	3 59	39
4 2 4 1	051	005	3 05	40
4 3 4 1	050	004	2 99	36
4 4 4 1	047	004	2 55	37
4 1 4 2	045	004	4 16	63
4 2 4 2	051	004	3 49	62
4 3 4 2	051	003	3 41	58
4 4 4 2	059	003	2 75	57

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TABLE 2.9 EQUIVALENT ORBITER DEADBANDS (WITH RESPECT TO THE IDEAL LOCAL VERTICAL ORIENTATION)

CASE NUMBER	DEADBANDS RATE (°/sec) PITCH	DEADBANDS RATE (°/sec) ROLL	DEADBANDS EXCURSION (degrees) PITCH	DEADBANDS EXCURSION (degrees) ROLL
1 1.0	0 06	-	-	-
1.1.2	0 06	-	2 0	-
1 1.3	-	-	-	-
1 2 1	0 05	-	-	-
1.2 2	0 05	-	3 0	-
1 2 3	-	-	-	-
1.3 1	0.06	-	-	-
1.3 2	0.06	-	3 0	-
1.3 3	-	-	-	-
1.4 1	0 05	-	-	-
1 4 2	0 05	-	2 0	
1 4 3	-	-	-	-
1 5 1	0 06	-	-	-
1 5 2	-	-	2 0	-
1.5 3	-	-	-	-
1 6 1	0 05	-	-	-
1 6 2	0 05	-	2 0	-
1 6.3	-	-	-	-
1 7 1	0 05	-	-	-
1 7 2	0 05	-	3 0	-
1 7 3	-	-	-	-
1 8.1	0.05	-	-	-
1.8 2	0.05	-	2 0	-
1 8 3	-	-	-	-

CASE NUMBER	DEADBANDS RATE (°/sec) PITCH	DEADBANDS RATE (°/sec) ROLL	DEADBANDS EXCURSION (degrees) PITCH	DEADBANDS EXCURSION (degrees) ROLL
2 1 1	0.06	-	-	-
2 1 2	0.06	-	2 0	-
2.1 3	-	-	-	-
2 2 1	0 05	-	-	-
2 2 2	0 05	-	3 0	-
2.2 3	-	-	-	-
2 3 1	0 05	-	-	-
2.3 2	0 05	-	3 0	-
2.3 3	-	-	-	-
2 4 1	0 05	-	-	-
2 4 2	0 05	-	3 0	-
2 4 3	-	-	-	-
2 5 1	0 06	-	-	-
2 5 2	0 06	-	2 0	-
2 5 3	-	-	-	-
2 6 1	0 06	-	-	-
2 6 2	0 06	-	2 0	-
2 6 3	-	-	-	-
2 7 1	0 05	-	-	-
2.7 2	0 05	-	3 0	-
2 7 3	-	-	-	-
2 8 1	0 05	-	-	-
2 8 2	0 05	-	2 0	-
2 8 3	-	-	-	-

CASE NUMBER	DEADBANDS RATE (°/sec) PITCH	DEADBANDS RATE (°/sec) ROLL	DEADBANDS EXCURSION (degrees) PITCH	DEADBANDS EXCURSION (degrees) ROLL
3.1.4.1	0 5	-	3.0	-
3.2 4 1	0 04	-	3 0	-
3 3 4.1	0 04	-	3 0	-
3 4 4 1	0 04	-	2 0	-
3 1 4 2	0 05	-	4 0	-
3 2.4 2	0 05	-	4 0	-
3 3.4 2	0 05	-	4 0	-
3 4 4 2	0 06	-	3.0	-
4 1 4 1	0 06	-	4 0	-
4 2 4 1	0 05	-	3 0	-
4 3 4 1	0 05	-	3 0	-
4 4 4 1	0 05	-	3 0	-
4 1 4 2	0 05	-	4 0	-
4 2 4 2	0 05	-	3 0	-
4 3 4 2	0 05	-	3 0	-
4 4 4 2	0 06	-	3 0	-

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Simulation Results

Viewing the data in Tables 2.3 through 2.6 and comparing to the requirements in Figure 2.9, the critical docking condition is lateral misalignment. Successful cases are indicated by the black triangles at the left. All the other docking conditions can be met easily, if a pitch deadband is maintained (HFRMP lacks the automatic capability to simulate limit-cycling deadband behavior). For Tables 2.3 and 2.4, nine cases are successful, 19% of the 9.1M (30 ft.) +$\overline{V}$ cases. For Table 2.4, 25% of the 15.2M (50 ft.) $\overline{R}$ cases and 50% of the 9.1M $\overline{R}$ cases are successful. Note the success rate is doubled by approximately halving the closure distance. This is to be expected as it gives the initial errors only half as long to propagate. The success rates of 9.1M for $\overline{V}$ and $\overline{R}$ approaches cannot be compared directly. Since the $\overline{V}$ cases had higher initial pitch rates, they needed more pitch corrections. Also, all the $\overline{R}$ cases used a single nose thruster to provide X_b thrust; 44% of the single nose thruster, pitch rotationally corrected 9.1M, 1.8 min., $\overline{V}$ cases were successful; 50% of the 1.8 min., 9.1M, $\overline{R}$ cases were successful. Interesting to note is the effect of one primary nose thruster (F3F) versus two (F F, F2F); only 8% of the two thruster $\overline{V}$ cases were successful as compared to 29% of the one thruster $\overline{V}$ cases. One thruster firing and missing its ideal time by .08 seconds gives a ΔV increment closer to the required value than two thrusters each missing their ideal time by .08 seconds.

All except one of the successful docking cases in Tables 2.3 and 2.4 occur when both the Z_b and X_b thrusters burn either 80 milliseconds too long or too short. These cases are virtually split between long or short. This makes sense because when one set of thrusters fires too long and the other too short, the error is effectively twice that of when they both fire short or long.

Approach attitude appears to have little effect in terms of aero drag. or gravity gradient, as long as the orbiter is in a stable or semistable attitude (one principal axis along $\overline{R}$ or $\overline{V}$ and one perpendicular to the orbit plane) because of the short docking times involved. This was proven early in the study by making identical HFRMP docking runs from 15.2M with and without full aero and gravity gradient effects. The only discernable effect was a difference in lateral deviation of 0.1 inches.

However, the offset of the c.g. from the orbiter docking port causes orbital mechanics effects, particularly in the sideways approach case (to be discussed later). The c.g. is displaced approximated 11.9M (39 ft) from the docking port along the X_b axis. Therefore, depending on the approach attitude, the ΔV requirements will be different because of the way the c.g. is oriented with respect to the docking port. The astronaut must be aware of this. Another c.g. related problem is orbiter rotation. The astronaut in the cabin cannot perceive a difference between orbiter translations and rotations because he is so far from the c.g. The solution to this would be to keep a very tight angle deadband. This approach was not desired during the simulation because the translations imparted during .08

sec. (minimum impulse) rotational corrections can make the lateral deviation worse. Also, the allowed 6 degree pitch misalignment for safe docking permits large deadbands.

Some deadband information can be gleaned from this study, however. Tables 2.8 and 2.9 show the maximum orbiter attitude errors and equivalent deadbands, respectively. Table 2.8 shows the largest rate and angle excursions per case from the initial terminal approach attitude. Yaw rates and angles are not shown because they are essentially zero. Notice how roll errors are very small, also, and probably do not require correction. The deadbands shown in Table 2.9 are derived by equating pitch corrections to deadbanding behavior. For applicable cases, a rounded-off value from Table 2.8 is listed in Table 2.9 as a rate deadband if it occurs at the beginning of the run and as an excursion deadband if it occurs halfway through the run. The roll values were so low that as before, they were excluded. The runs listed in Table 2.9 have rate deadbands of approximately .05°/sec., and excursion deadbands of roughly 3 degrees. If HFRMP had full attitude hold motion simulation capabilities (it can only simulate propellant used in attitude holds), the runs probably would not be significantly different if these deadbands were used. The rate deadband wouldn't be realistic, however, because the minimum rate deadband with primary jets, is 0.2 degree/second. Then, probably all attitude control during terminal approach would have to be accomplished using small (1.0 degree or less) pitch excursion deadbands. Vernier jets are capable of rate deadbands as low as .01 degree/second, but cannot be used because they are disabled when the primary mode is initiated.

An interesting effect occurred on all runs because of the ΔV component imparted in the $+X_b$ direction when firing the $-Z_b$ direction primaries. On runs with initial attitudes set up so that the tail ($+X_b$) primary thrusters would provide the impulse velocity needed along the X_b axis (+V, nose down for example), the Z_b primary thruster firing yielded more impulse along $+X_b$ than necessary. Therefore, the nose primaries had to fire, instead of the tail primaries, to decrease the impulse along the X_b axis to the correct level. The same happened in all other cases, because the initial attitude was selected so that the orbiter docking port faced the SOC. Most of the ΔV for idealized docking is required in the $-Z_b$ direction, therefore, the $+X_b$ component of thrust due to a $-Z_b$ command builds up due to the relatively long firing time in the $-Z_b$ direction. Figure 2.11 illustrates this for V docking, with one and two nose thrusters, versus terminal docking distance. The time to dock has been normalized out of the plotted data by requiring an axial docking speed of about 0.1M/sec (.33 ft/sec). At approximately 16.1M (53 ft) for the nose down orbiter the X_b thrusters do not have to be used because the $-Z_b$ thrusters contribute enough $+X_b$ thrust. For greater distances they do not provide enough thrust (or ΔV impulse) so for nose down cases the aft thrusters must be used. For a nose up orbiter, the $-Z_b$ thrusting provides $+X_b$ impulse in the opposite direction to that required, so the nose thrusters must always null out this ΔV component besides providing the ΔV of opposite sign for docking. This is why the nose up case curve never crosses the horizontal. The curves for other approach cases are about the same.

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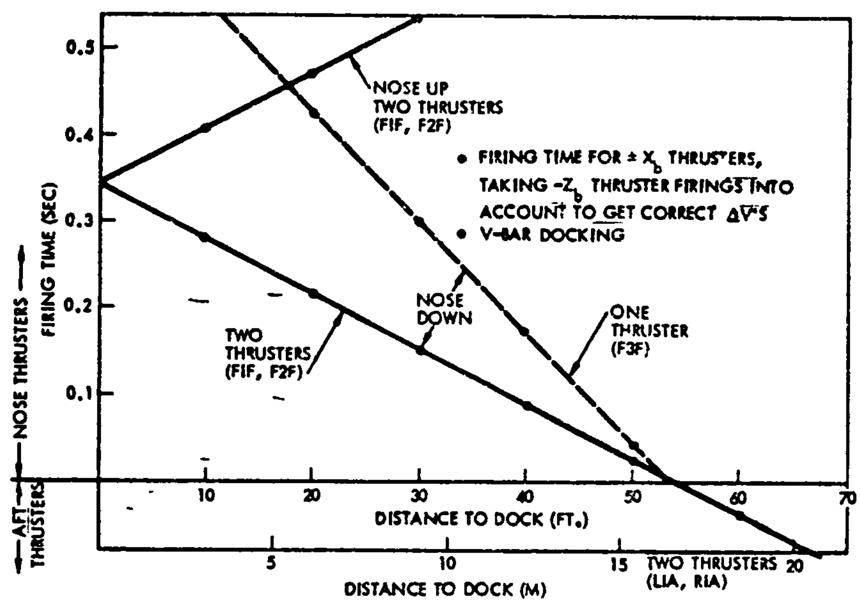

FIGURE 2.11 REQUIRED FIRING TIMES FOR X_B PRIMARY THRUSTERS

The sideways $\overline{V}$ docking cases are presented in Table 2.6. They are presented separately because they were not analyzed as thoroughly as in-plane $\overline{V}$ and $\overline{R}$ cases and because the procedure involved in sideways V docking requires an extra out-of-plane translational burn and extra rotational burns. The orbiter is initially sideways with its docking port lined up in the SOC's orbital plane and with its relative velocity in the X direction initially zero. This requires an orbit with a slightly different inclination for the orbiter because the c.g. is not in the same plane as SOC.

Two in-plane burns are required, ($-Z_b$, $+Y_b$) identical to those required for a tail-up or tail-down $\overline{V}$ approach. However, a third out-of-plane burn ($-X_b$) is necessary to line up the docking port at contact. Because of a lack of thruster compensation, the third burn is not independent of the first two. Three axis translational thrust coupling causes especially high rotational rates. A human operator would have difficulty in predicting the coupled reactions to a selected translational command. Unlike $\overline{V}$ and $\overline{R}$ in-plane maneuvers which require only pitch corrections, sideways docking requires corrections about all axes, inducing additional errors. Table 2.6 shows that at 9.1M, sideways docking is worse than $+\overline{V}$ docking at 15.2M (Table 2.5).

The main conclusion from this analysis is that the orbiter can be flown to successful docking contact conditions, but that "close in" ΔV adjustments, mostly in the X and Y directions will be required. Also, the character and number of these corrective impulses will be affected by the closing geometry $\bar{V}$, $\bar{R}$, etc., and orbiter attitude, tail up or down, X-POP, etc.

Although the above simplified simulation analysis did not include realtime man-in-the-loop response characteristics the general conclusions appear valid in light of the crew controlled terminal closure simulations conducted several years ago. Man-in-the-loop simulations with high fidelity orbiter flight control characteristics were conducted by JSC in support of LDEF retrieval investigations. The objectives of this simulation activity are summarized in Figure 2.12 along with some of the conclusions judged to be pertinent to the current SOC study. The results identified problem sensitivities to approach path and orbiter attitudes similar to those described above. However, piloting techniques were developed which indicated the capability to fly up to and stationkeep with the coorbiting LDEF target vehicle to within 0.009MPS (0.03 FPS) relative velocity in all axes. This is well within the docking contact velocity envelope specified for SOC and, thus, confirms the belief that the orbiter can successfully dock with the SOC under normal conditions.

STUDY APPROACH PATH OPTIONS TO LDEF, JSC - 12776, NOV 7, 1977

OBJECTIVES:

- ORBITER CAPABILITY TO APPROACH & STATIONKEEP WITH PLUME SENSITIVE P/L
- ESTABLISH PERFORMANCE DATA BASE FOR VARIOUS APPROACH TECHNIQUES

RESULTS:

- $\bar{R}$ & $\bar{V}$ APPROACHES OFFER OPERATIONAL ADVANTAGES
- $\bar{H}$ APPROACHES ARE CONDITIONALLY FEASIBLE
- ALL APPROACH MODES SHOULD BE CONSIDERED
- PILOTS CONSISTANTLY MAINTAINED STATIONKEEPING WITHIN 0.03 FPS ALL AXES
- SPECIAL TECHNIQUES CAN GREATLY REDUCE PLUME EFFECTS ±X PRCS ("LOW Z") BRAKING
- FORE & AFT CCTV's ARE VALUABLE AIDS FOR FINAL CLOSURE

A TOTAL OF 20 CONCLUSIONS & RECOMMENDATIONS ON TECHNIQUES, TRENDS, & SENSITIVITIES RELATIVE TO PLUME EFFECTS & FUEL CONSUMPTION

FIGURE 2.12 JSC TERMINAL CLOSURE SIMULATION RESULTS

2.2.4 RCS Jet Failure Considerations

While it is believed that the orbiter can dock with the SOC under normal conditions, as discussed above, the possibility of off-nominal situations must also be considered. Perhaps the most severe of these is related to RCS jet failures, particularly the failed on or "runaway jet" condition. The potential for a single point failure which can cause a runaway jet has been identified. The reaction jet drivers (RJD) which are shown in the flight control system schematic in Figure 2.2 can fail in such a way as to cause an RCS jet to fire. The RJDs provide "jet on" commands to the individual RCS jets in response to DAP and hand controller signals. Thus, an electrical short in a RJD can result in a single point runaway jet failure. While a failure alarm sounds almost immediately to alert the crew, the current software is not able to automatically diagnose which jet is firing. It is estimated that up to one minute of crew action may be required to identify the failed jet and take corrective action. Thus, the critical zone for a runaway jet is just before docking contact is made.

If a runaway jet occurs in this zone it may result in inadvertent contact between the orbiter and the SOC outside of the docking capture envelope. This "critical closure zone" is analogous to the critical flight zone of a helicopter as illustrated in Figure 2.13. If the helicopter is

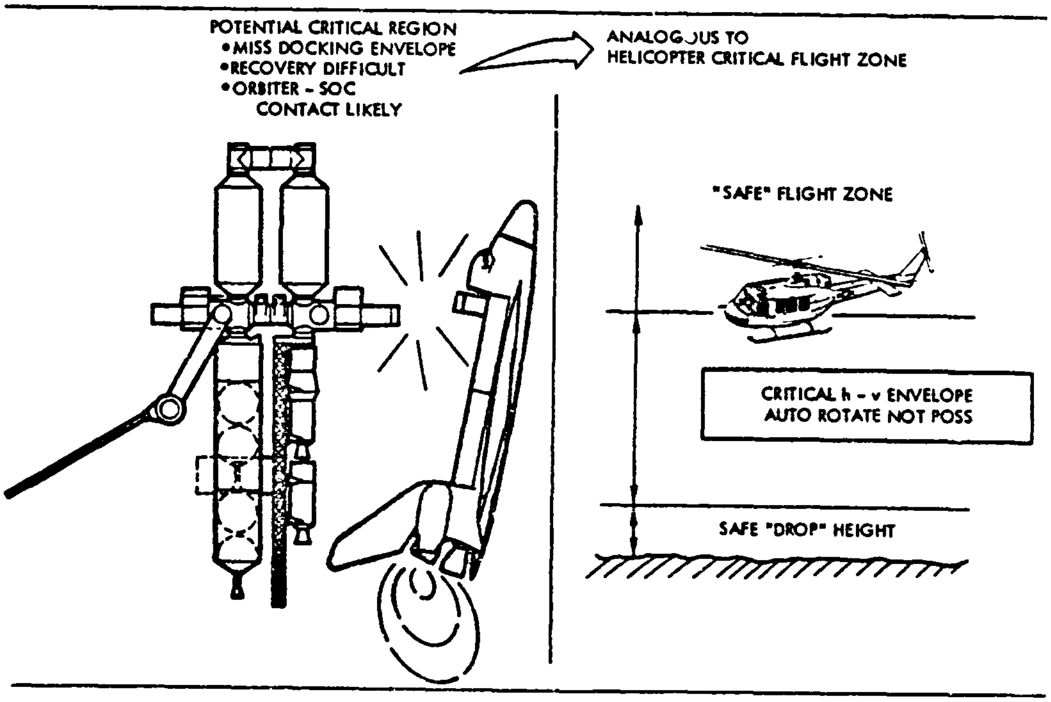

FIGURE 2.13 Critical Zone Analogy

sufficiently low when the failure occurs it can drop safely to the ground in a survivable hard landing. Also, if its altitude is sufficiently high when the failure occurs a successful autorotation and flare maneuver to a safe landing can be performed. However, there is a middle or critical zone in which a free fall drop is high enough to be fatal and yet is too low for successful autorotation. Similarly, in the SOC docking operation there may be a zone where there is insufficient stopping distance to perform a safe turnaround abort maneuver prior to contact.

Although this problem can only be fully analyzed with high fidelity man-in-the-loop simulations a simplified analysis of the trajectory deviations and abort actions was conducted to assess the general severity of the problem. It was concluded that critical closure zones are likely to exist for some recovery techniques, but the overall problem is not as critical as was initially feared.

<u>Runaway Jet Problem Model Description</u>

The important problem elements covering the reference configuration and system geometry factors are summarized in the following paragraphs.

The nominal orbiter/SOC docking scenario is illustrated in Figure 2.14. It is assumed that the docking maneuver utilizes a "tailchase" V-bar approach (along the orbital velocity vector) in a tail down attitude and that the primary crew aids are the overhead optical sight (COAS) and video displays from orbiter mounted cameras. It is also assumed that the orbiter is placed in a local vertical hold (LVH) rotational control mode with the orbiter attitude coaligned to the SOC attitude in a port-to-port matchup for docking. This will reduce the pilot workload by freeing his attention from rotational control while concentrating on the closing translation maneuvers. There are two important assumptions within this docking scenario which are pertinent to the runaway jet analyses. They are: (1) the initial relative motion is exactly along the orbiter Z-axis toward the SOC, and (2) the two docking ports are perfectly aligned at the onset of the jet failure.

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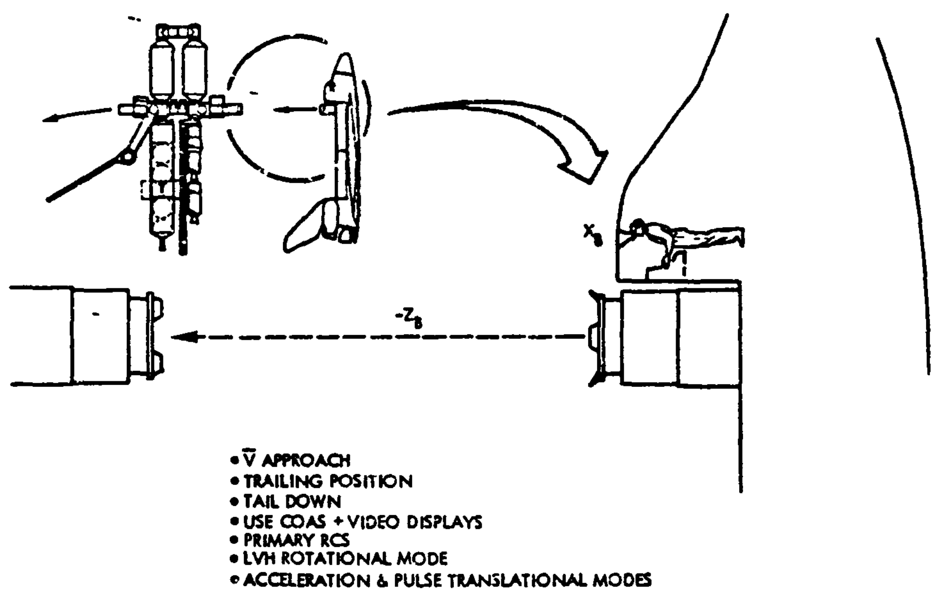

FIGURE 2.14 ORBITER/SOC NOMINAL DOCKING SCENARIO

The geometrical arrangement within the STS coordinate system and the pertinent dimensional and mass property characteristics used in the problem model are shown in Figures 2.15 and 2.16. The X and Z displacements of the docking port from the orbiter c.g. will result in significant linear motions due to coupling with angular rotations.

There are 44 primary and vernier RCS thrusters on the orbiter. The individual thruster identification codes together with plume and induced motion directions are pictured in previous Figure 2.4. Thrust components in body axes along with the grid coordinates of the point of thrust application are listed in Table 2.1 of Section 2.2.2.

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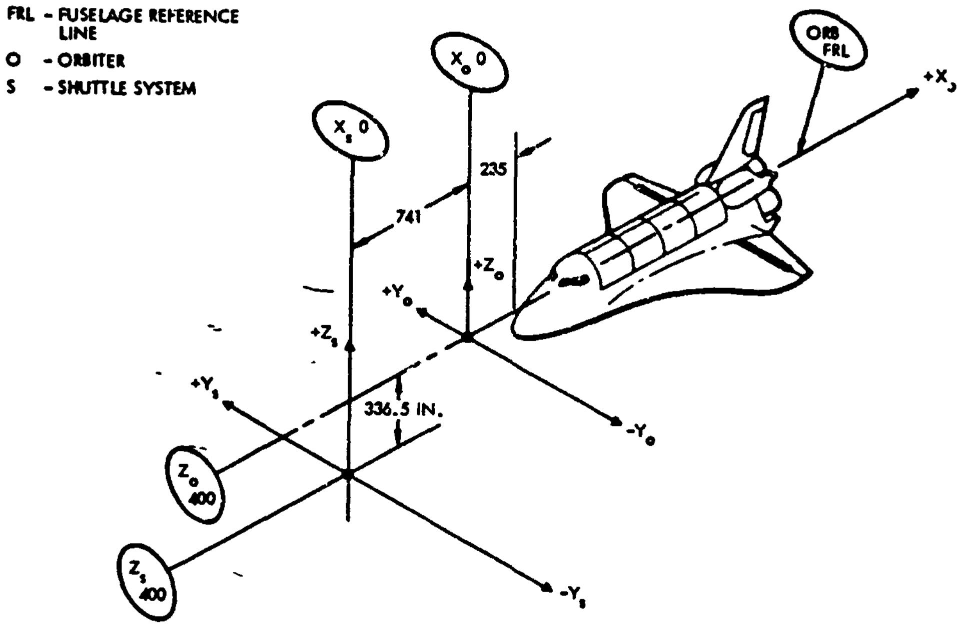

FIGURE 2.15 SHUTTLE COORDINATE SYSTEMS

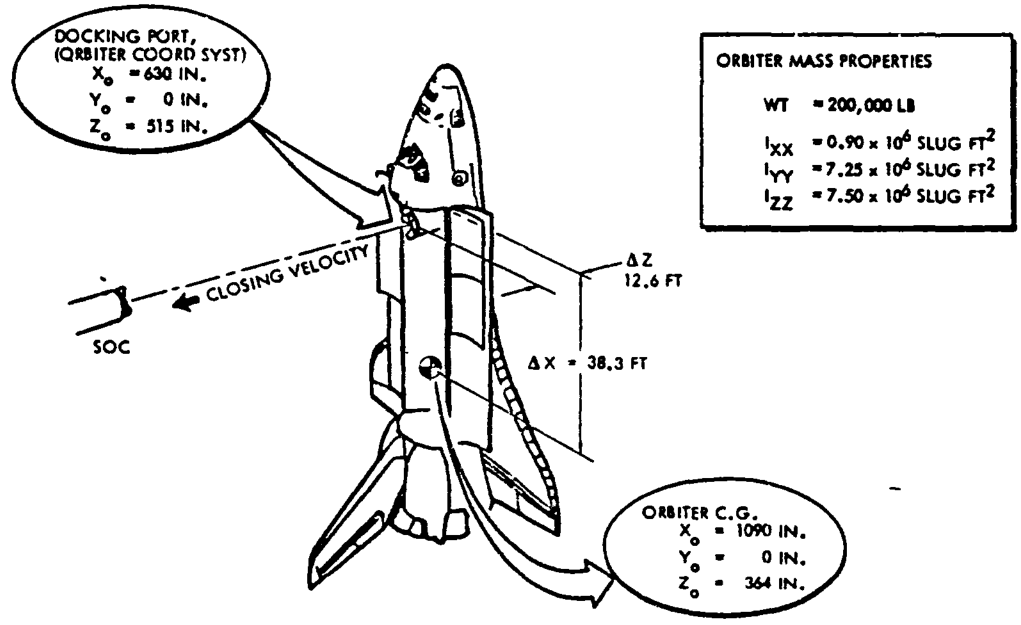

FIGURE 2.16 RUNAWAY JET MASS MODEL

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Another important element in the problem model is the digital autopilot (DAP). It governs the pattern of RCS thruster firings in response to the various control commands. The specific jet select logic which is mechanized into the DAP software is summarized in Tables 2.10, 2.11, and 2.12. The number of jets to fire from each of the 14 directional groups is shown for all possible commands including pure rotation, pure translation and combined rotation and translation. The different jet response patterns associated with single and dual failures (failed-off) are also indicated in the tables. Table 2.10 shows all the normal Z thrusting modes and related rotational commands. Included are the special rotational modes, "low nose" and "low tail" in which only the nose or tail jets are used to perform rotational control. The "normal" pure rotational control mode is depicted in the middle row of entries with the caption Normal. The bottom two rows cover combined rotational control and +Z translation and -Z translation respectively. The row of numbers (4, 5, 6, etc.) across the caption bars at the top of the table indicate the thruster groups (1 through 14) as defined in Figure 2.4 and the entires below in the body of the table indicate how many jets will fire in each group for the various command combinations.

TABLE 2.10 NORMAL Z JET SELECT LOGIC

		NUMBER OF JETS TO FIRE IN EACH GROUP																																		
		NO FAILURES, SINGLE FAILURES, AND MOST DUAL FAILURES							DUAL FAILURE EXCEPTIONS																											
									2 FAILURES IN GP 5							2 FAILURES IN GP 6							2 FAILURES IN GP 13							2 FAILURES IN GP 14						
CONDITION	COMMANDS	4	5	6	11	12	13	14	4	5	6	11	12	13	14	4	5	6	11	12	13	14	4	5	6	11	12	13	14	4	5	6	11	12	13	14
NORMAL Z OR NO Z TRANSLATION	LOW NOSE																																			
	+PITCH	0	1	1	2	0	0	0	0	0	1	0	0	0	0	0	1	0	0	0	0	0				–							–			
	-PITCH	1	0	0	0	0	0	0				–							–							–							–			
	+PITCH/+ROLL	0	0	0	0	1	0	0				–							–							–							–			
	+PITCH/-ROLL	0	0	0	1	0	0	0				–							–							–							–			
	-PITCH/+ROLL	0	0	0	0	0	2	0				–							–				0	0	0	0	0	1	0				–			
	-PITCH/-ROLL	0	0	0	0	0	0	2				–							–							–				0	0	0	0	0	0	1
	LOW TAIL																																			
	+PITCH	0	0	0	1	1	0	0				–							–							–							–			
	-PITCH	0	0	0	0	0	1	1				–							–							–							–			
	+PITCH/+ROLL	0	0	0	0	1	0	0				–							–							–							–			
	+PITCH/-ROLL	0	0	0	1	0	0	0				–							–							–							–			
	-PITCH/+ROLL	0	0	0	0	0	2	0				–							–				0	0	0	0	0	1	0				–			
	-PITCH/-ROLL	0	0	0	0	0	0	2				–							–							–				0	0	0	0	0	0	1
	NORMAL																																			
	+PITCH	0	1	1	1	1	0	0	0	0	1	1	1	0	0	0	1	0	1	1	0	0				–							–			
	-PITCH	1	0	0	0	0	1	1				–							–							–							–			
	+ROLL	0	0	0	0	1	1	0				–							–							–							–			
	-ROLL	0	0	0	1	0	0	1				–							–							–							–			
	+PITCH/+ROLL	0	1	1	0	1	0	0	0	0	1	0	1	0	0	0	1	0	0	1	0	0				–							–			
	+PITCH/-ROLL	0	1	1	1	0	0	0	0	0	1	1	0	0	0	0	1	0	1	0	0	0				–							–			
	-PITCH/+ROLL	1	0	0	0	0	2	0				–							–				1	0	0	0	0	1	0				–			
	-PITCH/-ROLL	1	0	0	0	0	0	2				–							–							–				1	0	0	0	0	0	1
	+Z	1	0	0	1	1	0	0				–							–							–							–			
	+PITCH	0	0	0	1	1	0	0				–							–							–							–			
	-PITCH	1	0	0	0	0	0	0				–							–							–							–			
	+ROLL	1	0	0	0	1	0	0				–							–							–							–			
	-ROLL	1	0	0	1	0	0	0				–							–							–							–			
	+PITCH/+ROLL	0	0	0	0	1	0	0				–							–							–							–			
	+PITCH/-ROLL	0	0	0	1	0	0	0				–							–							–							–			
	-PITCH/+ROLL	1	0	0	0	1	0	0				–							–							–							–			
	-PITCH/-ROLL	1	0	0	1	0	0	0				–							–							–							–			
	-Z	0	1	1	0	0	2	2	0	0	1	0	0	2	2	0	1	0	0	0	2	2	0	1	1	0	0	1	1	0	1	1	0	0	1	1
	+PITCH	0	1	1	0	0	0	0	0	0	1	0	0	0	0	0	1	0	0	0	0	0				–							–			
	-PITCH	0	0	0	0	0	1	1				–							–							–							–			
	+ROLL	0	1	1	0	0	2	0	0	0	1	0	0	2	0	0	1	0	0	0	2	0	0	1	1	0	0	1	0				–			
	-ROLL	0	1	1	0	0	0	2	0	0	1	0	0	0	2	0	1	0	0	0	0	2				–				0	1	1	0	0	0	1
	+PITCH/+ROLL	0	1	1	0	0	2	0	0	0	1	0	0	1	0	0	1	0	0	0	1	0	0	1	1	0	0	1	0				–			
	+PITCH/-ROLL	0	1	1	0	0	0	2	0	0	1	0	0	0	1	0	1	0	0	0	0	1				–				0	1	1	0	0	0	1
	-PITCH/+ROLL	0	0	0	0	0	2	0				–							–				0	0	0	0	0	1	0				–			
	-PITCH/-ROLL	0	0	0	0	0	0	2				–							–							–				0	0	0	0	0	0	1

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Table 2.11 shows the firing patterns for the high-Z translation mode and Table 2.12 presents the X and Y axis translation cases with pertinent combinations of yaw rotational commands. In general these tables are independent. Each shows the response to unique commands, which can then be added to determine the combined thruster firing patterns for mixed motions such as $\dot{Z}$ and $\dot{Y}$ plus desired rotations. The above problem geometries and jet response logic were applied to simplified analyses of runaway jet effects.

TABLE 2.11 HI -Z JET SELECT LOGIC

		NUMBER OF JETS TO FIRE IN EACH GROUP														
		NO. OF FAILURES *			SINGLE FAILURE EXCEPTIONS: ONE FAILURE –GRP 4			ONE FAILURE –GRP 11			ONE FAILURE –GRP 12			ALL DUAL FAILURES		
CONDITION	COMMANDS	4	11	12	4	11	12	4	11	12	4	11	12	4	11	12
HIGH +Z WITH (CLOSED XFD OR INTERCONNECT) NO +X TRANSLATION	HIGH +Z	3	3	2	2	2	2	3	2	2		–		1	1	1
	+PITCH	2	3	3		–		1	2	2		–		0	1	1
	-PITCH	3	1	1	2	1	1		–			–		1	0	0
	+ROLL	3	1	2	2	1	3		–		2	0	2	1	0	1
	-ROLL	3	3	1	2	3	1	2	2	0		–		1	1	0
	+PITCH/+ROLL	1	1	3		–			–		0	0	2	0	0	1
	+PITCH/-ROLL	1	3	1		–		0	2	0		–		0	1	0
	-PITCH/+ROLL	3	0	2	2	0	2		–			–		1	0	1
	-PITCH/-ROLL	3	2	0	2	2	0		–			–		1	1	0
HIGH +Z WITH (OPEN XFD OR INTERCONNECT) +X TRANSLATION	HIGH +Z/+X	3	2	2	2	2	2		–			–		1	1	1
	+PITCH	1	2	2		–			–			–		0	1	1
	-PITCH	3	1	1	2	1	1		–			–		1	0	0
	+ROLL	3	1	3	2	1	3		–		2	0	2	1	0	1
	-ROLL	3	3	1	2	3	1	2	2	0		–		1	1	0
	+PITCH/+ROLL	1	1	3		–			–		0	0	2	0	0	1
	+PITCH/-ROLL	1	3	1		–		0	2	0		–		0	1	0
	-PITCH/+ROLL	3	0	2	2	0	2		–			–		1	0	1
	-PITCH/-ROLL	3	2	0	2	2	0		–			–		1	1	0

*MOST SINGLE FAILURES

TABLE 2.12 X AND Y JET SELECT LOGIC

	NO. JETS TO FIRE IN EA. GROUP											
	NO. OF FAILURES*				DUAL FAILURE EXCEPTIONS							
					2 IN GP 2				2 IN GP 3			
COMMANDS	2	3	9	10	2	3	9	10	2	3	9	10
LOW TAIL												
+ YAW	0	0	0	0			–				–	
- YAW	0	0	1	0			–				–	
LOW NOSE												
+ YAW	1	0	0	0	0	0	0	1			–	
- YAW	0	1	0	0			–		0	0	1	0
NORMAL												
+ YAW	1	0	0	1	0	0	0	1			–	
- YAW	0	1	1	0			–		0	0	1	0
+Y	1	0	1	0	0	0	0	0			–	
-Y	0	1	0	1			–		0	0	0	0
+Y/+YAW	1	0	0	0	0	0	0	1			–	
+Y/-YAW	0	0	1	0			–				–	
+Y/+YAW	0	0	0	1			–				–	
+Y/-YAW	0	1	0	0			–		0	0	1	0

*SINGLE AND MOST DUAL FAILURES

	NO OF FAILURES *			NO. JETS TO FIRE IN EA. GRP								
				DUAL FAILURE EXCEPTIONS								
				2/GRP 1			2/GRP 7			2/GRP 8		
COMMAND	1	7	8	1	7	8	1	7	8	1	7	8
+X	0	1	1		–		0	0	1	0	1	0
-X	2	0	0	1	0	0		–			–	

*SINGLE AND MOST DUAL FAILURES

Runaway Jet Analysis

The simplified analysis presented here mainly looks at the trajectory deviations resulting from runaway jet actions and assesses the ability of the orbiter to apply corrective thrust actions within the control modes and operating constraints judged to apply to the docking scenario. First, the general factors that can affect the critical closure zone are:

- o Relative approach speed
- o Approach path accuracy (deviations at the time of failure)
- o Pilot reaction time in identifying and responding to the emergency
- o The amount of corrective thrust authority

Key assumptions and/or implications on the role of these factors in the subsequent analysis are summarized in the accompanying discussion.

The approach speed is governed somewhat by the allowed docking contact conditions which can be up to 0.15MPS (0.5 FPS) for the SOC. Slightly lower speeds would likely be used in actual docking operations to allow for errors on the fast side. The use of much lower approach speeds would greatly increase the time required for docking and would increase the effects of orbit mechanics and gravity gradient perturbations. Thus, for conservatism (it makes the abort problem more difficult) a value of 0.15MPS was used here.

Trajectory accuracy affects the amount of the safe docking envelope which can be allocated to runaway jet effects. If substantial devaitions exist at the time of jet failure even less maneuvering room may be available for recovery than for the ideal case. However, there are no accurate simulation results upon which deviation estimates can be based. Thus, for expediency it was assumed that no path deviations and/or attitude misalignments existed at the time of failure. This may not be as overly optimistic as imagined because the region of interest is very close to the point of contact and the deviations should be small.

Crew reaction time will be affected by the conditions and nature of the failure and by operating procedures yet to be developed. They can only be fully evaluated by real time man-in-the-loop simulations. However, some of the fundamental effects of different reaction times can be estimated by indexing certain results in the trajectory deviation data.

The corrective thrust authority is essentially dictated by the DAP jet select logic and the RCS thruster geometry. These, coupled with assumed input commands representative of various response options were used to evaluate the general capability of the orbiter to counter runaway jet effects.

Only the 38 primary RCS thrusters need be considered in the runaway jet analysis because the verniers cannot be used during the docking translation maneuvers. They would be deactivated for this flight phase. As indicated earlier these 38 thrusters are organized into 14 directional groups, but because of left-right symmetry there are only eight unique translation/rotation motion conditions which can be caused by a runaway jet. These eight motion conditions and the corresponding corrective thruster response selections from the jet select logic, Tables 2.10 through 2.12, are summarized in Table 2.13. The corrective thrust response selections represent only the combination of thruster firings which will counter the effects of the runaway jet. These would be indicative of a strategy which calls for continuation of the docking maneuver while trying to counter the effects of the runaway jet.

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TABLE 2.13 UNIQUE MOTIONS FROM RUNAWAY JETS

	RESULTING MOTION		JETS TO COUNTER RESULTING MOTION*
X-AXIS JETS			
GP-1	+PITCH	$-X_B$	(1) GP-7 AND (1) GP-8
GP-7 & GP-8	-PITCH	$+X_B$	(2) GP-1
Y-AXIS JETS			
GP-2 } SYMMETRICAL, LEFT & RIGHT	+YAW	$+X_B$	(1) GP-3**
GP-3 } SYMMETRICAL, LEFT & RIGHT	-YAW	$-Y_B$	(1) GP-2**
GP-9 } SYMMETRICAL, LEFT & RIGHT	-YAW	$+Y_B$	(1) GP-10**
GP-10 } SYMMETRICAL, LEFT & RIGHT	+YAW	$-Y_B$	(1) GP-9**
Z-AXIS JETS			
GP-4	-PITCH	$+Z_B$	(1) GP-5 AND (1) GP-6
GP-11 & -12	+PITCH	$+Z_B$	(1) GP-13 AND (1) GP-14
GP-13 & -14	-PITCH	$-Z_B$	(1) GP-11 AND (1) GP-12
GP-5 & -6	+PITCH	$-Z_B$	(1) GP-4**

*FROM ON-ORBIT DAP JET SELECT LOGIC LOOK-UP TABLES
**MOTION FROM RUNAWAY JET CANNOT BE COMPLETELY ARRESTED

Other strategies or procedures could be employed which call for abort actions to reverse the closing velocity and then back the orbiter off to a safe distance. Abort procedures both with and without attempts to simultaneously correct for the deviations induced by the runaway jet could be attempted. A number of these response options were investigated including variations in what parameter, what event or what times were used to "trigger" the corrective action.

First, to assess the basic stopping power of the orbiter in performing an abort maneuver the stopping times and distances were determined as a function of closing velocity. These are shown in Figure 2.17 and include both the normal-Z and high-Z translation thrusting modes. They further reflect the different numbers of Z thrusters which can result from the combined effects of rotational control, translation thrusting and the continuing action of the runaway jet.

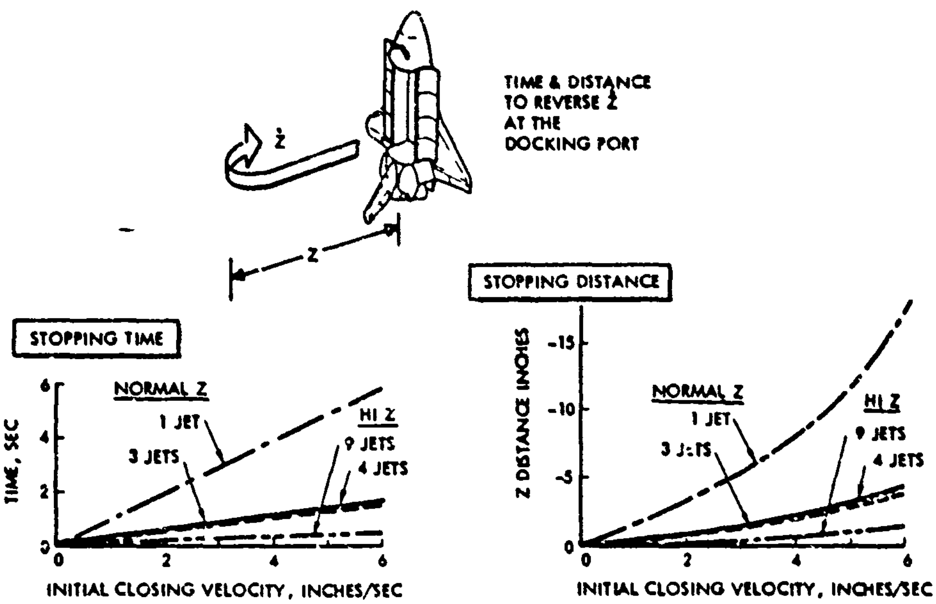

FIGURE 2.17 DOCKING ABORT TURNAROUND

An example case covering a runaway jet in group 14, an aft Z-thruster on the right hand side producing thrust in the minus Z_B direction, is shown in Figure 2.18. The main motions induced by this failure are the minus Z_B acceleration combined with pitch down and roll left angular disturbances. The corresponding jet firings to counter these effects from the DAP logic lookup tables are shown for both the normal-Z and high-Z modes. As indicated in the figure, the normal-Z case calls for a single thruster firing from Group 12. This action combined with the opposing thrust from the runaway thruster results in a net stopping force of zero. For the high-Z case the corrective thrust response calls for one thruster each from Groups 4 and 11 and three thrusters from Group 12 for a total of five thrusters. The net stopping force, accounting for the runaway jet is thus equivalent to four Z thrusters.

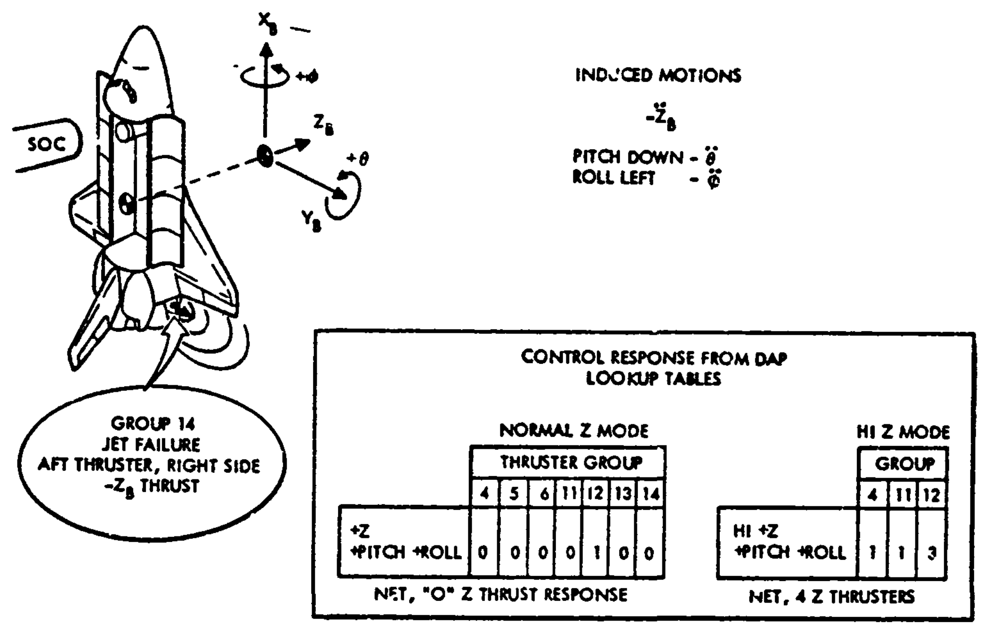

FIGURE 2.18 EXAMPLE RUNAWAY JET SCENARIO

This type of stopping power assessment was conducted for all eight unique motion conditions and gives the range of jet numbers for the stopping time and distance curves back in Figure 2.17. For most situations the normal-Z thrusting response calls for the equivalent of from one to three thrusters net stopping force while that for the high-Z mode ranges from four to nine thrusters.

X-Thruster Deviations

The stopping times for these ranges of abort response characteristics, based on a 0.15 MPS closing rate, were superimposed on trajectory deviation plots to determine the amount of deviation which could occur before the abort stopping action could be completed. The results for runaway X thrusters are shown in Figure 2.19. The dominant deviation parameters are ΔX, ΔZ and $\Delta\Theta$ (pitch angle). These deviation data were calculated at the docking port and include the effects of both translation and orbiter rotation. As indicated by these curves, a single runaway X-jet with no corrective thrust applied drives the closing path outside of the safe docking envelope ($\pm$.23M (9 in) radial offset) within approximately three seconds. The stopping time for the normal-Z thrusting mode is approximately four seconds for Groups 7 or 8 failures and nearly six seconds for a Group 1 failure. The ΔX displacement for these cases ranges from .38M (15 in) to .72M (30 in) well outside of the allowable $\pm$ .23M (9 in) envelope.

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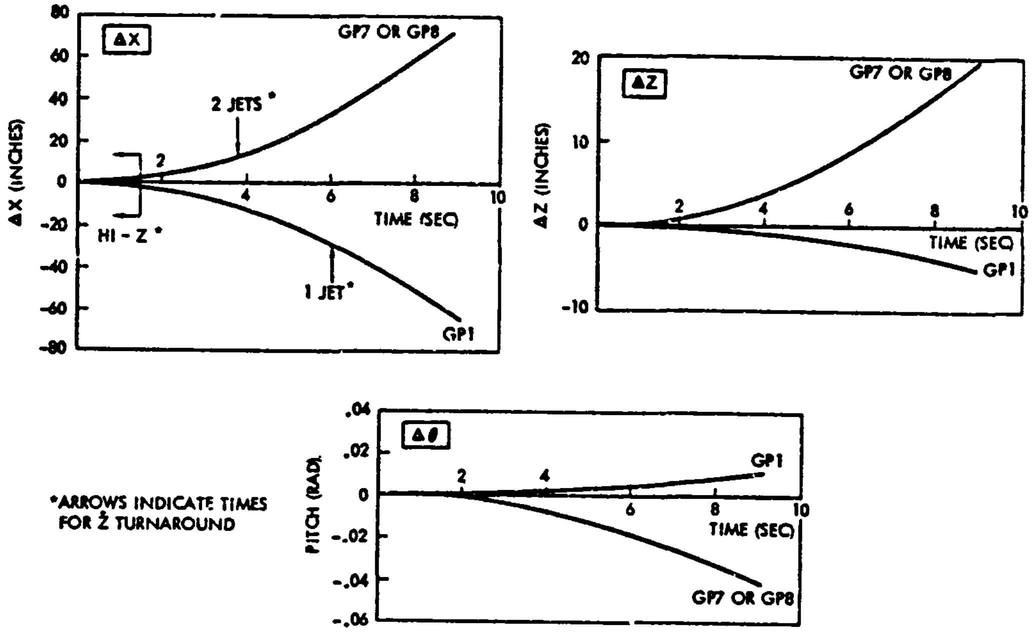

FIGURE 2.19 DEVIATIONS FROM X RUNAWAY JETS

However, for the high-Z abort turnaround mode the stopping time is approximately 1.5 seconds which reduces the ΔX deviations to approximately .05M (2 in) or .08M (3 in). Thus, if the runaway jet should occur at, say, two seconds before normal docking contact even without corrective action the docking contact would occur within ± .23M envelope. If the runaway jet should occur earlier than the two second point, the high-Z mode provides sufficient stopping power to reverse the closing motion before contact is made. Thus, the high-Z mode virtually eliminates the possibility of contact outside of the docking envelope due to runaway X-thrusters if the strategy to always abort in the event of a runaway jet is adopted.

Other procedures were also briefly explored. Figures 2.20 through 2.25 illustrate various triggering criteria and corrective maneuver assumptions for a runaway jet in group 1. They are based on the following: reaching an angle deadband limit, specific times from the instant of failure and different deviation distance limits. These are all indicative of crew "eyeball" responses to preceived motions. The dominant motions from a Group 1 jet firing are negative X acceleration with a positive pitch (nose up) rotation.

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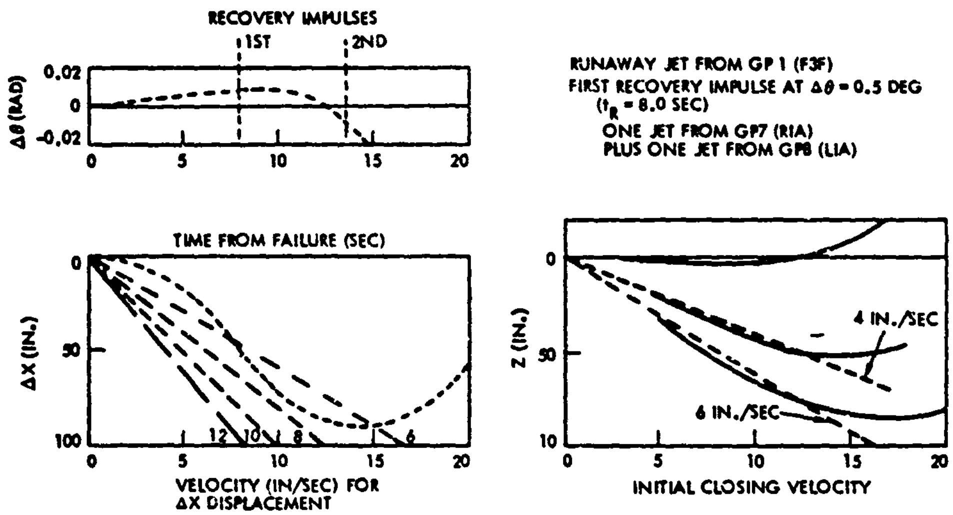

FIGURE 2.20 MOTION FROM GROUP 1 JET FAILURE, CORRECTION INITIATED AT Δθ = 0.5 DEG.

Several corrective triggering assumptions were investigated together with "pure" deviation correction maneuvers (no abo t action). First, a correction maneuver was assumed to be initiated when the attitude deadband of 0.5 degrees is reached. If the flight control system is in a hold attitude mode it will automatically initiate compensating rotational thrust at this point. It was arbitrarily assumed for this trajectory run that control thrusting for plus X deviations would also be introduced at this point. With these assumptions a critical closure zone (contact outside of the ± .23M envelope) will exist if the docking maneuver has not been completed within 3.5 seconds after thruster failure. The motion characteristics including the combined effects of the runaway jet and the corrective thruster firing are illustrated in Figure 2.20. The critical closu e zone as a function of initial closing velocity is shown in Figure 2.21.

Cases where corrective maneuvers are initiated by lateral displacement (deviation) limits are presented in Figures 2.22 and 2.23. When the recovery thrust is applied at ΔX = 12.7 cm (5 inches), which is at approximately 2-2.5 seconds, the critical closure zone can be completely eliminated. The use of abort maneuvers triggered by the five inch deviation

• CORRECTION INITIATED AT $\Delta\theta$ = 0.5 DEG DEADBAND LIMIT

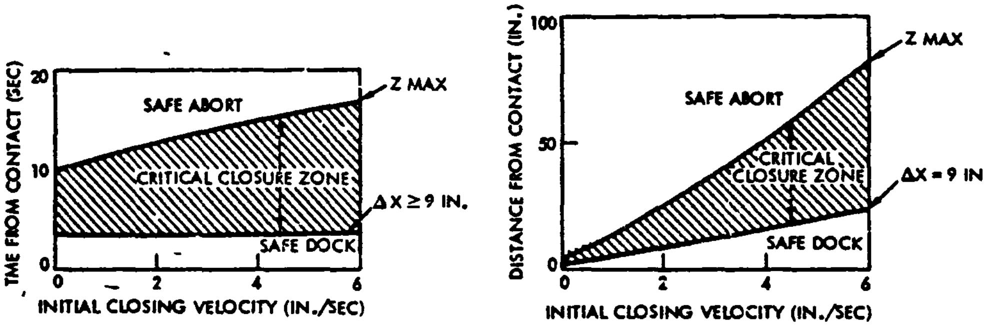

FIGURE 2.21 CRITICAL CLOSURE ZONE FOR GROUP 1 JET FAILURE

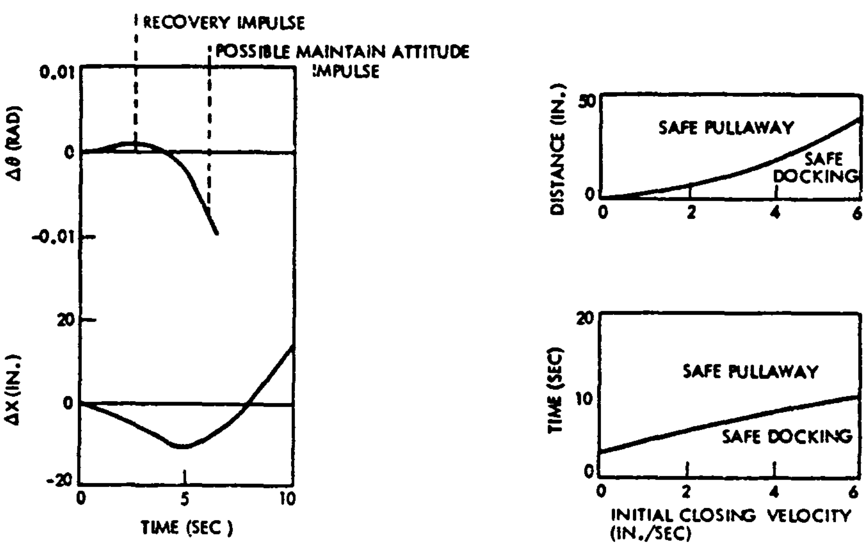

FIGURE 2.22 MOTION FROM GROUP 1 JET FAILURE,
CORRECTION INITIATED AT ΔX = 5 IN.

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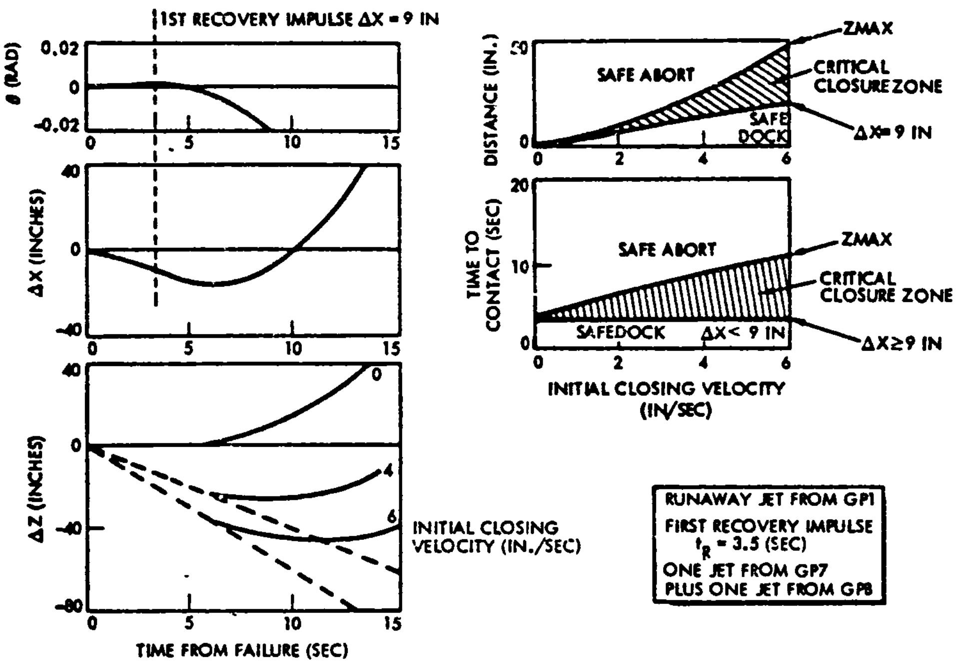

FIGURE 2.23 MOTION FROM GROUP 1 JET FAILURE,
CORRECTION INITIATED AT ΔX = 9 IN.

criteria were also investigated. In Figure 2.24 the abort maneuver is combined with deviation corrections. The "abort only" procedure is illustrated in Figure 2.25. Both of these procedures provide essentially the same turnaround capability. For the abort only mode (Figure 2.25) the results are summarized below for both the normal-Z and high-Z modes.

Thrust Mode	Begin Recovery Time	Turnaround Conditions	
		time	Δx
Normal Z	2 sec	3.8 sec	27.9 cm (11 inches)
Hi Z	2 sec	2.8 sec	16.5 cm (6.5 inches)

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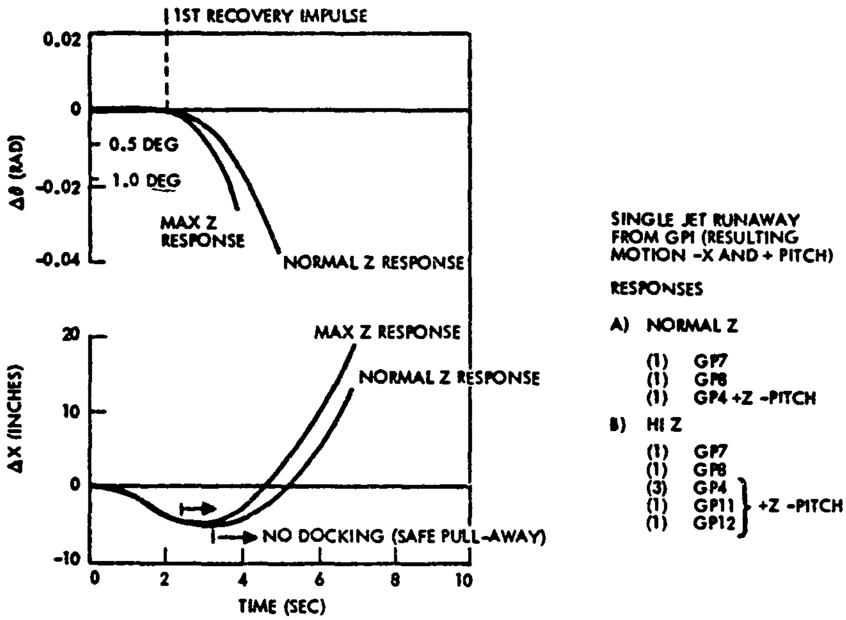

FIGURE 2.24 GROUP 1 RUNAWAY JET WITH ABORT PLUS DEVIATION CORRECTIONS

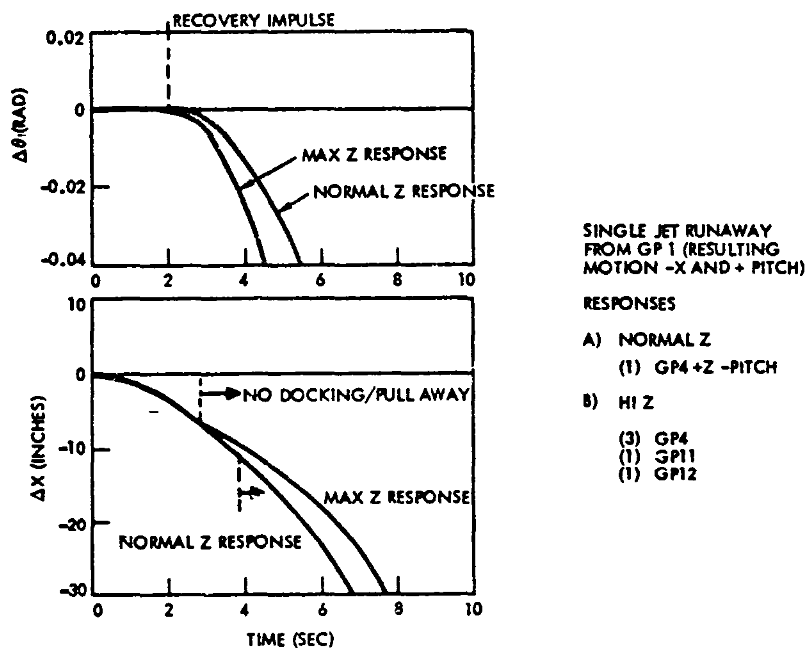

FIGURE 2.25 GROUP 1 RUNAWAY JET WITH ABORT ONLY CORRECTIONS

The trajectory histories for docking aborts with a runaway jet in either Groups 7 or 8 are presented in Figures 2.26 and 2.27. The main results are again summarized as follows:

Thrust Mode	Begin Recovery Time	Turnaround Conditions	
		time	Δx
Normal Z	2 sec	>10 sec	>75 cm (>30 inches)
Hi Z	2 sec	2.5 sec	14 cm (5.5 inches

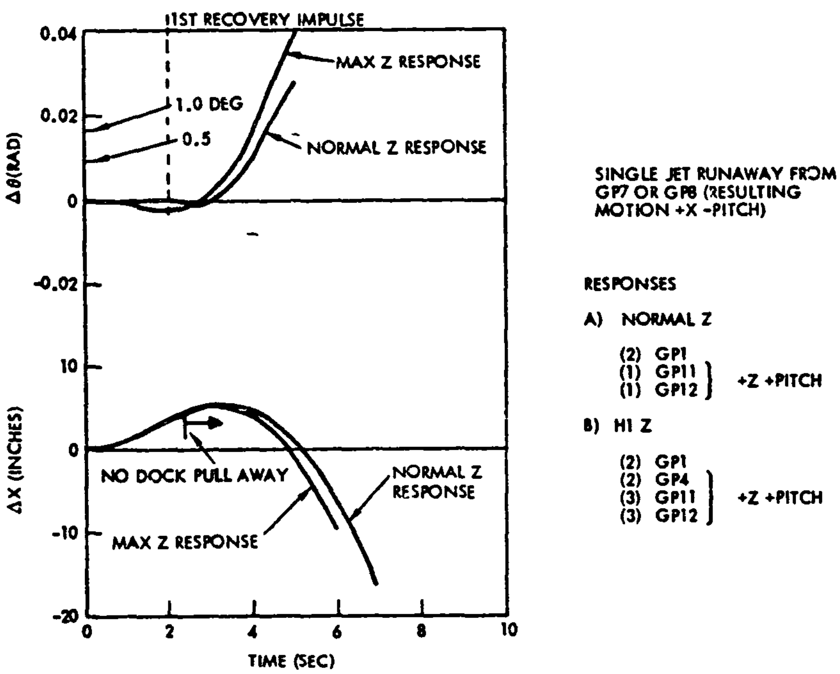

FIGURE 2.26 GROUP 7 OR 8 RUNAWAY JET WITH ABORT PLUS DEVIATION CORRECTIONS

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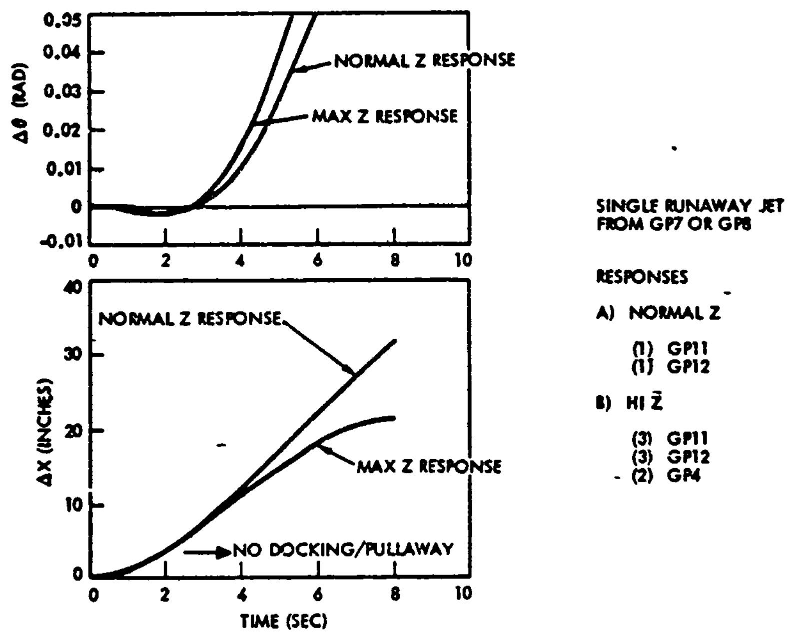

FIGURE 2.27 GROUP 7 OR 8 RUNAWAY JET WITH ABORT ONLY CORRECTIONS

Y-thruster Deviations

The dominant trajectory deviations due to runaway Y-thrusters (Groups 2 and 3 forward and Groups 9 and 10 aft) are shown in Figure 2.28. From the DAP jet select logic (Table 2.12) it can be seen that all combinations of $\pm$Y translation thrusting with $\pm$ yaw commands result in single thruster firings which in the presence of a runaway jet produce a net stopping force of zero. There is no high-thrust mode for Y translation. At best, only the Y acceleration can be nulled and the Y velocity accumulated up to the point of response cannot be arrested. This effect is clearly shown in Figures 2.29 and 2.30 where even with corrective thrusting the ΔY deviation continues to increase linearly.

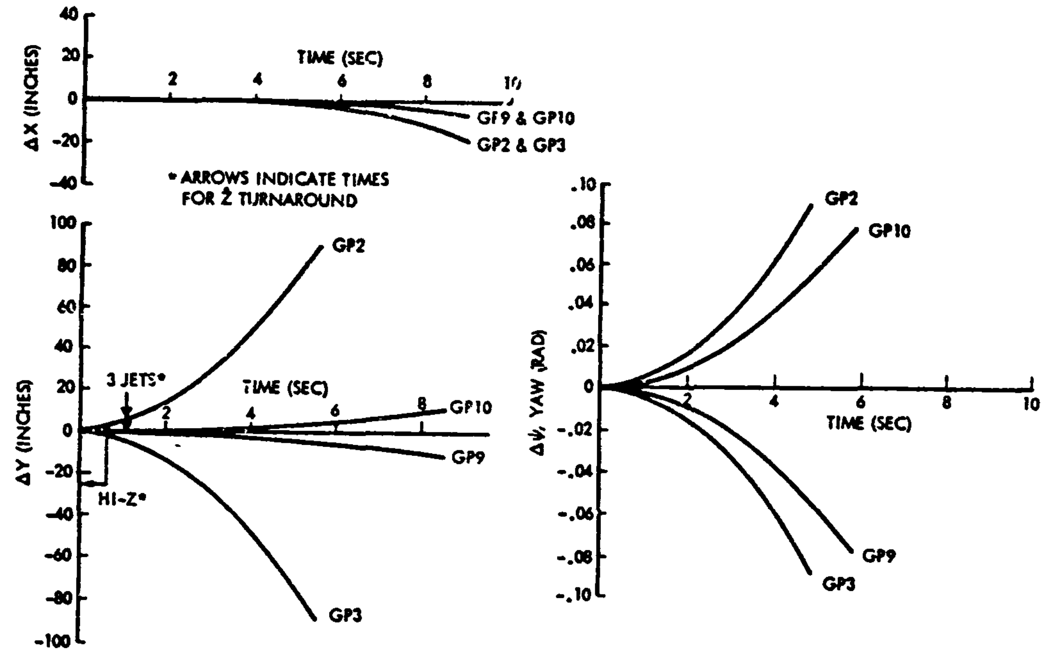

FIGURE 2.28 DEVIATIONS FROM Y RUNAWAY JETS

The only viable procedure for runaway Y thrusters is to activate the Z jets and abort the docking maneuver. The "stopping times" for arresting a docking approach velocity of 0.15 MPS are about 0.5 and 1.0 seconds for high-Z and normal-Z modes respectively. As was the case with X runaway jets

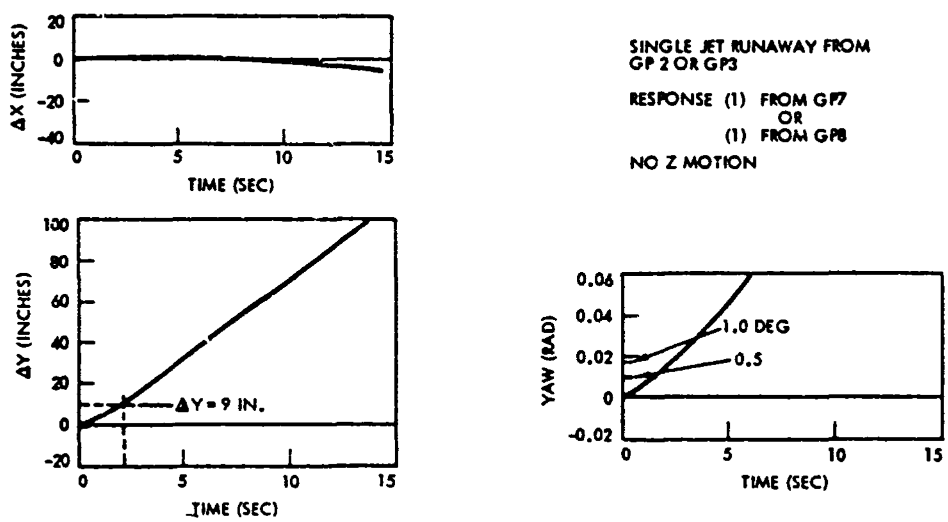

FIGURE 2.29 GROUP 2 OR GROUP 3 RUNAWAY JET

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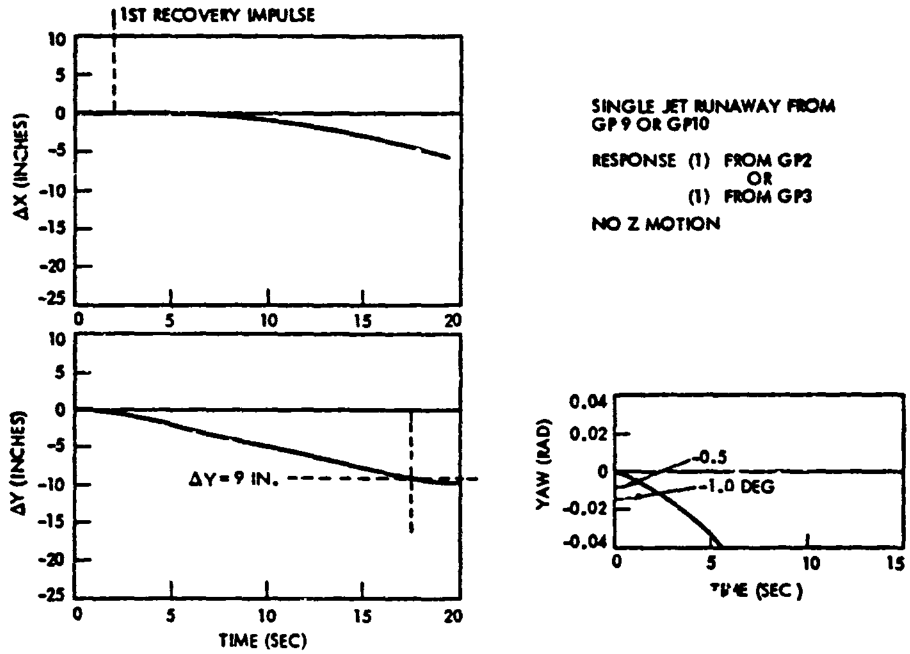

FIGURE 2.30 GROUP 9 OR GROUP 10 RUNAWAY JET

the "high-Z" abort mode appears to provide satisfactory margins to preclude contact outside of the ± .23M envelope. Only three or four inches of ΔY deviation (approximately 9 cm) can occur within the high-Z stopping time. On the other hand, the normal-Z mode (three jets) is marginal for Groups 2 or 3 failures, particularly if allowances are made for initial path deviations and pilot response time in reacting to the failure. Deviations of 8 or 9 inches (21 cm) are possible within the normal-Z mode stopping times.

Z-Thruster Deviations

The dominant trajectory deviations due to runaway Z-thrusters (Groups 4, 5 and 6 forward and Groups 11, 12, 13 and 14 aft) are illustrated in Figure 2.31. Since the Z-thrusters also affect pitch motion there is a considerable amount of ΔX deviation at the docking port.

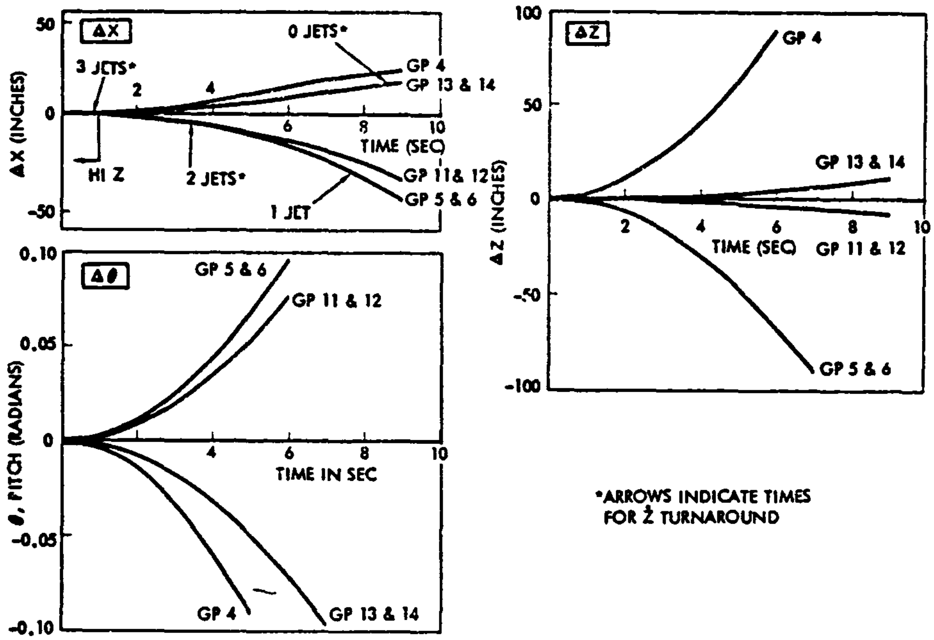

FIGURE 2.31 DEVIATIONS FROM Z RUNAWAY JETS

Recovery trajectory histories for the four unique Z-thruster deviation conditions are presented in Figures 2.32 through 2.35. The recoveries are initiated essentially at two seconds from the failure onset. Both normal and high-Z responses are shown. With the exception of normal-Z aborts from Groups 13 and 14 failures, sufficient thrust authority exists to either complete the docking maneuver of abort safely. Groups 13 and 14 failures result in "zero net stopping force with the normal-Z mode.

Since there is no immediate indication of which thruster has failed, the safest procedure for any runaway jet during docking is probably to abort to a safe standoff distance using the high-Z mode, identify and shutoff the failed jet, and then redo the docking operation.

Thus, in all cases, X, Y or Z runaway jet failures, the "high-Z" translation thrust mode appears to offer sufficient stopping power to successfully abort the docking maneuver without inadvertant contact outside the safe docking envelope.

Failed "Off" Jet

A failed off jet is judged not to be a serious problem. There is sufficient redundancy in the primary RCS system that no basic thrust authority is lost with any single failed off jet. This can be easily seen in the thruster identification code of Figure 2.4, which shows there are two to four thrusters in each of the 14 directional groups. This redundancy is further exhibited in the jet select logic tables (Tables 2.10, 2.11 and 2.12) where all of the RCS firing schedules for the various command combinations are unaffected by a single failure. Hence, all maneuvers can be achieved without degradation. The only effect will be to introduce a 0.24 second time delay (three DAP sampling intervals, 80 milliseconds each) in the first thrusting action after the failure. The DAP then switches to the priority two jet in the affected jet group and all thrusting actions thereafter call for the use of this jet and no further 0.24 second time delays occur.

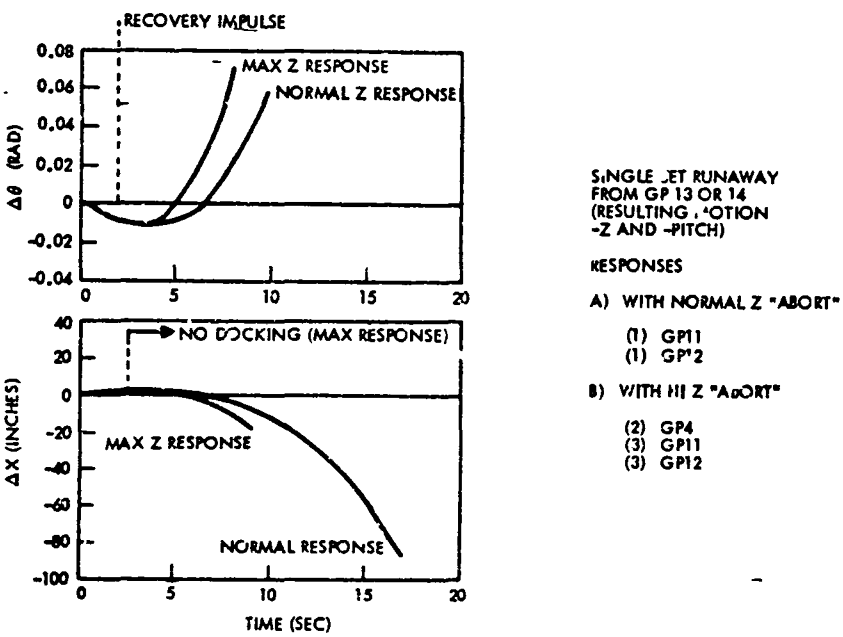

FIGURE 2.32 RUNAWAY JET FROM GROUP 13 OR 14, ABORT ONLY CORRECTIONS

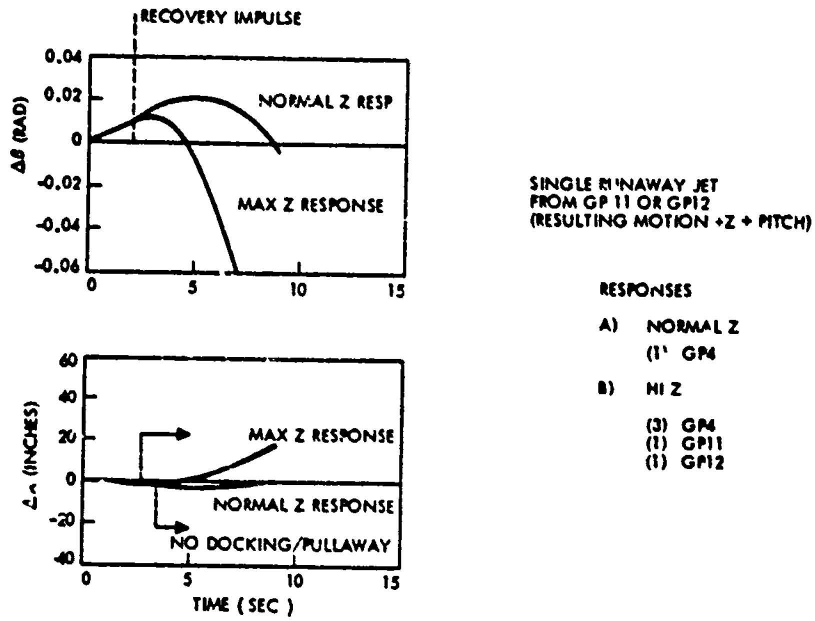

FIGURE 2.33 RUNAWAY JET FROM GROUP 11 OR 12, ABORT ONLY CORRECTIONS

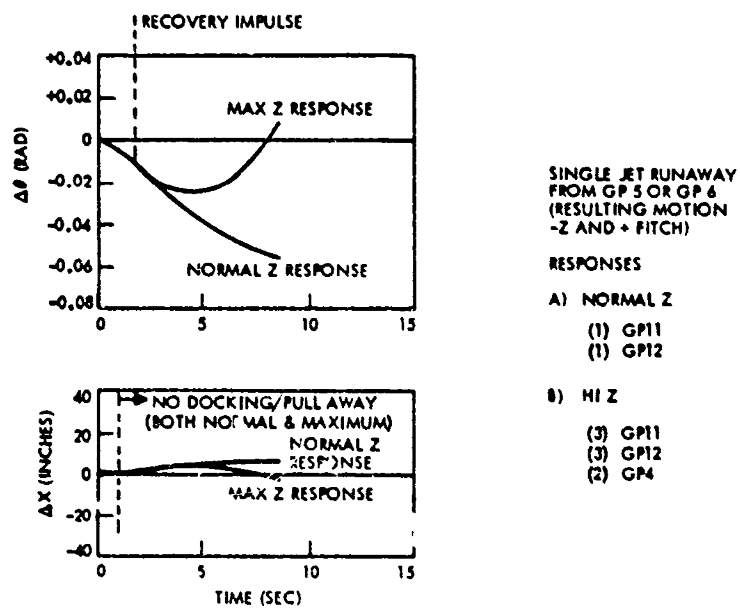

FIGURE 2.34 RUNAWAY JET FROM GROUP 4, ABORT ONLY CORRECTIONS

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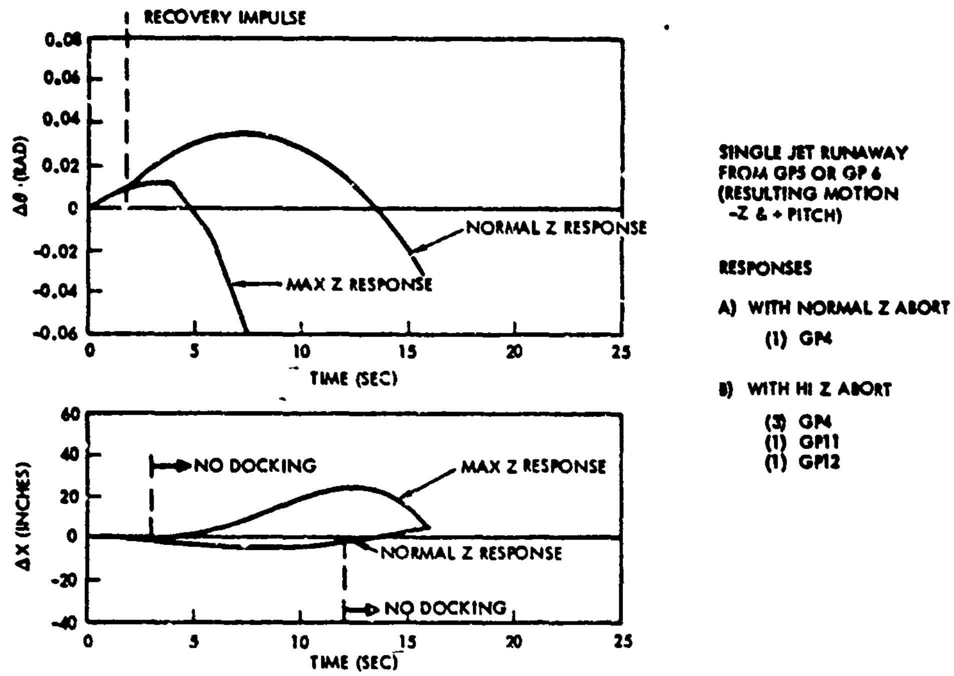

FIGURE 2.35 RUNAWAY JET FROM GROUP 5 OR 6, ABORT ONLY CORRECTIONS

2.2.5 Plume Impingement

The final factor to be considered in the overall analysis of proximity operations is the plume impingement environment created by the orbiter during the final docking maneuvers. It was previously determined that a worst case situation would occur when "High-Z" thrusting is used to abort the docking approach. The specific problem model for this condition is defined and the resulting analysis approach and results are presented in the following paragraphs.

<u>Problem Statement</u>

Determine the resulting forces and moments onto the SOC as a result of plume impingement from the orbiter RCS engines. Assume that all nine +Z-firing primary engines and one Y-firing primary engine in the forward pod are firing, see Figure 2.3 for pictorial representation of the orbiter RCS. The orbiter is assumed to be almost mated to the SOC and the solar array panel facing the Y-thruster that is operating is at an angle of 52° to its boom. In addition, the solar array panel frontal area is at its maximum with respect to the line-of-sight of the operating Y-facing thruster. Assume the SOC center-of-gravity is located on the centerline of the SOC/orbiter mating boom approximately 11.7 M (460 inches) from the SOC/orbiter interface.

Plume Flow Model Assumptions

For this analysis, only line-of-sight surfaces were considered as being affected by the exhaust plume. Thus, reflected flow phenomena (secondary effects) were ignored. All SOC surfaces were assumed to be in the free-molecular (far field) flow regime with respect to the RCS thrusters. This assumption can be proven valid for all surfaces except the front of the service module that mates with the orbiter. Consequently, the free-molecular flow model used for the orbiter RCS primary thruster was an axisymmetric, source flow approach with nozzle boundary layer effects included. The axisymmetric assumption eliminated any consideration of nozzle scarfing effects in the flow model. To include nozzle scarfing effects, would require three-dimensional equations rather than the two-dimensional equations used. In addition, the flux attenuates in the free-molecular flow regime as the square of the distance from the origin, i.e., the nozzle exit plane/ nozzle centerline intersection.

Also, because of the far-field assumption, the effects of several plumes onto the same surface can be superimposed. Further, the three 870-lbf engines from any one engine pod can be replaced by a pseudo-single 2610-lbf engine whose mass flux, momentum flux, and energy flux were three times larger than one 870-lbf engine and whose location is where the middle 870-lbf thruster is located.

Working Flux Equations

Mass:

$$\dot{m}_T = \int_0^{X_{max}} \left(\frac{d\dot{m}}{d\Omega}\right) (2\pi \sin X \, dX) \qquad \text{(Eq. 1)}$$

where

$$\ell_n \left(\frac{d\dot{m}}{d\Omega}\right) = A_0 + A_2X^2 + A_3X^3 + A_4X^4 + A_5X^5 \qquad \text{(Eq. 2)}$$

Force:

$$F_T = \int_0^{X_{max}} V_\ell \left(\frac{d\dot{m}}{d\Omega}\right) [2\pi \sin X \cos X \, dX] \qquad \text{(Eq. 3)}$$

Mass distribution flux:

$$\frac{d\dot{m}}{d\Omega} = \rho V R^2 \qquad \text{(Eq. 4)}$$

Momentum distribution flux:

$$\frac{V_\ell}{g_c} \left[\frac{d\dot{m}}{d\Omega}\right] = \rho V^2 R^2 \qquad \text{(Eq. 5)}$$

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Energy distribution flux:

$$\frac{V\ell^2}{2Jg_c}\left[\frac{d\dot{m}}{d\Omega}\right] = \rho V^3 R^2 \qquad \text{(Eq. 6)}$$

where:

A_0	Zero order term in mass flux equation	= ℓ_n 2.9422
A_2	Second order term in mass flux equation	= -1.7164
A_3	Third order term in mass flux equation	= -5.1523
A_4	Fourth order term in mass flux equation	= 5.2734
A_5	Fifth order term in mass flux equation	= -1.3091
F_T	Total Force	= 870 lbf
g_c	Gravitational constant	= 32.17 lbm·ft/lbf-sec^2
J	Energy Conversion constant	= 778 ft-lbf/BTU
$\dot{m}_T$	Total Mass	= 3.0 1 lbm/sec
R	Distance from source flow origin to impingment pt, ft	
V_ℓ	Limiting Velocity	= 11,448.5 ft/sec
X	Flow angle in radians	(X_{max} = 2π/3)

The curves in Figures 2.36 through 2.38 were generated using Equations 4, 5 and 6 for an 870-lbf thruster. However, to obtain the final mass flux equation, engine operating and plume data from References 1 through 4 were used in conjunction with the far-field plume mapping techniques described in References 1 and 6. Subsequent to curve-fitting the mass distribution flux (Eqs. 1 and 2, to match the flow rate of 3.0104 lbm/sec, it was determined that a limiting velocity (Vℓ) of 11,448.5 ft/sec produced a resultant force of 870-lbf using Eq. 3. This value compared favorably with the average limiting velocity of 11,564.4 ft/sec obtained from Reference 1. However, the calculated ideal limiting velocity based on parameters presented in Table 1 of References 2 and 4 gave a value of 10,805 ft/sec. For this analysis, it was decided to use the derived velocity of 11,448.5 ft/sec. By so doing, the results would be consistant with the given thrust.

The energy flux was calculated based on the assumption that the relationship presented in Eq. 6 was valid. In this presentation, the energy flux is used as the basis to obtain the convective heat transfer rate. This technique should give an upper bound; however, to obtain a more representative convective heating rate (or force per unit area), Reference 6 should be used.

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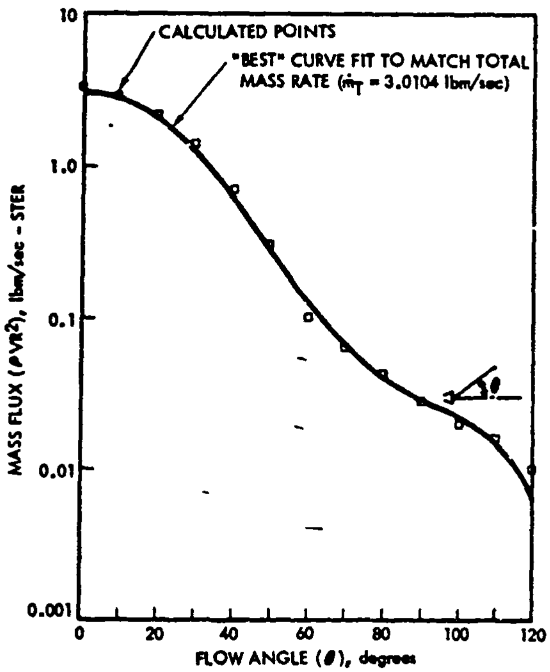

$$\dot{m}_T = \int_0^X \left(\frac{d\dot{m}}{d\Omega}\right) d\Omega$$

$$d\Omega = 2\pi \sin X \, dX$$

$$\ell n\left(\frac{d\dot{m}}{d\Omega}\right) = A_0 + A_2X^2 + A_3X^3 + A_4X^4 + A_5X^5$$

$A_0 = \ell n\ 2.9422$
$A_2 = -1.7164$
$A_3 = -5.1523$
$A_4 = 5.2734$
$A_5 = -1.3091$

X = FLOW ANGLE IN RADIANS

FIGURE 2.36 MASS FLUX FOR AN 870-LB BIPROPELLANT THRUSTER AS A FUNCTION OF FLOW ANGLE

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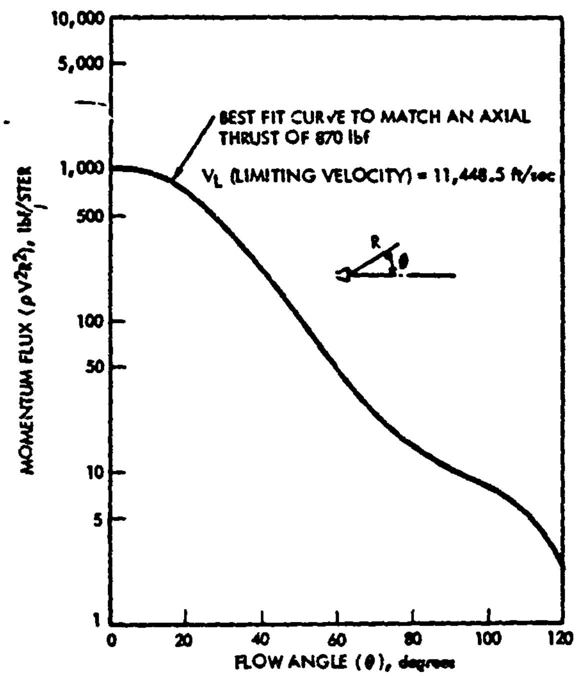

FIGURE 2.37 MOMENTUM FLUX FOR AN 870-LB BIPROPELLANT THRUSTER AS A FUNCTION OF FLOW ANGLE

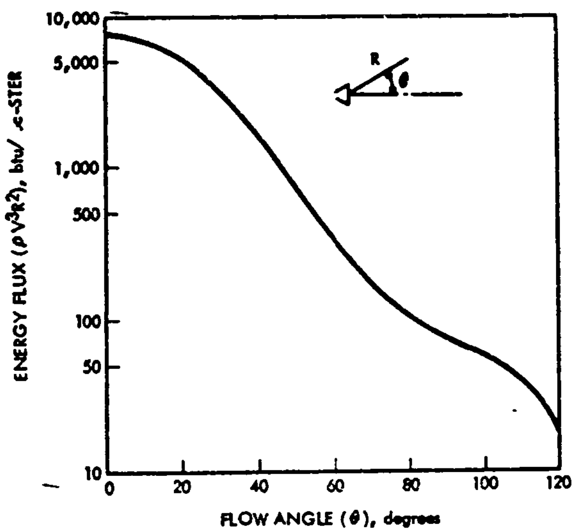

FIGURE 2.38 ENERGY FLUX FOR AN 870-LB BIOPROPELLANT THRUSTER AS A FUNCTION OF FLOW ANGLE

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Sample Problem

Assume that the three +Z-thrusters in the forward pod are firing onto the Habitability Module No. 1 from a distance of 8.9 M (350 inches), nozzle exit plane to nearest surface. Find the mass flux, pressure, and corrective heating rate along the nozzle centerline of the center engine onto the habitat module surface (assume a flat plate normal to the streamline).

1. $\theta = 0$
2. $R = 350/12 = 29.167$ ft
3. $d\dot{m}/d\Omega = 3$ lbm/sec-ster (Figure 2.36)
4. $\dot{m} = 3/R^2 = 3/(29.167)^2 = 3.53 \times 10^{-3}$ lbm/sec-ft^2 (one engine)
5. By assumption, $3 \times \dot{m} = \dot{m}$ (three engines) $= 0.0106$ lbm/sec-ft^2
6. $er^2R^2 = 1000$ lbf/ster (Figure 2.37)
7. $P = 1000/(29.167)^2 = 1.175$ psf $= 8.163 \times 10^{-3}$ psi (one engine)
8. By assumption $3 \times P = P$(three engines) $= 0.0245$ psi
9. $E = 7600$ BTU/sec-ster (Figure 2.38)
10. $\dot{Q} = 7600/(29.167)^2 = 8.93$ BTU/ft^2-sec (one engine)
11. By assumption, $3 \times \dot{Q} = \dot{Q}$ (three engines) $= 26.8$ BTU/ft^2-sec

For this sample problem, the answer would be that the mass flux onto the module surface from the three thrusters is 0.0106 lbm/sec-ft^2. the pressure is 0.0245 psi and the convective heating rate is 26.8 BTU/ft^2-sec. Whether these effects are harmful or not cannot be determined with the information given.

Discussion Concerning Flux Equations

The characteristics of the mass distribution in the free molecular flow regime is considered as good as, if not superior to, any available data set for the 870-lbf bipropellant thruster. Consequently, the determination of the mass flux is considered to be accurate. Because the mass distribution flux is well represented, the resultant momentum flux is considered to be a fair representation. Discussions may arise concerning the direct application of the energy flux distribution to determine convective heating rates. It is generally accepted that this approach gives an upper limit; however, the discussion arises concerning the derived value. It may be too conservative since it does not consider surface-to-gas temperature differences, energy accommodation coefficient, and the gas and Reynolds numbers. For this analysis, the approach will be conservative.

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Plume Pressure and Energy Equations

The following simplified hypersonic normal and tangential pressure equations and convective heat transfer equations were used in this analysis (see Reference 6 for equation development):

$$P_N = (2 - \alpha_n)\ \rho V^2 \sin^2 B \qquad \text{(Eq. 7)}$$

$$P_T = \alpha_T\ \rho V^2 \sin B \cos B \qquad \text{(Eq. 8)}$$

$$Q = \frac{1}{2} \alpha_e\ \rho V^3 \sin B \qquad \text{(Eq. 9)}$$

where

P_N	Pressure normal to the impinged surface, force/unit area
P_T	Pressure tangential to the impinged surface, force/unit area
Q	Convective heat transfer rate, heat rate/unit area
α_e	Energy accommodation coefficient
α_n	Normal momentum accommodation coefficient
α_T	Tangential momentum accommodation coefficient
B	Angle between surface and incident velocity vector
ρ	Gas density, lbm/ft^3
V	Limiting velocity, ft/sec

Substituting the momentum and energy flux equations and assuming the accommodation coefficients equal to unity, the following working equations were derived:

$$P_N = \left[\frac{V_\ell}{R^2 g_c} \cdot \frac{d\dot{m}}{d\Omega} \right] \sin^2 B \qquad \text{(Eq. 10)}$$

$$P_T = \left[\frac{V_\ell}{R^2 g_c} \cdot \frac{d\dot{m}}{d\Omega} \right] \sin B \cos B \qquad \text{(Eq. 11)}$$

$$\dot{Q} = \left[\frac{V_\ell^2}{2R^2 J g_c} \cdot \frac{d\dot{m}}{d\Omega} \right] \sin B \qquad \text{(Eq. 12)}$$

Note that when the angle of incidence (B) is normal to the surface, the equations are identical to the flux method used in the sample problem.

Computer Program

Rockwell personnel have used the pressure and convective heating rate equations (Eqs. 10, 11 and 12) in several of its desk top computer programs for other thruster and spacecraft configurations. Simply stated, the surface in question is subdivided into small increments. Subsequently, the flux and indident angle to the controls of each element are determined and the resultant pressures and heating rates per unit area are determined using Equations 10, 11, and 12. The pressures and heating rates per unit area are then multiplied by the elemental area, thus determining forces and heating rates on the elemental area. The resultant torques from each element are calculated to the known system center-of-gravity. To determine the overall forces, heating rates, and vehicle torques resulting from the plume impinging onto the surface, all the incremental forces, heating rates, and torques are summed.

Effects On SOC Hardware Elements

Shown in Figures 2.39, 2.40 and 2.41 is the SOC/orbiter configuration used in the plume impingement analysis. The SOC modules and elements potentially affected by the operation of the plus Z-firing RCS thrusters and one minus Y-firing thruster are summarized in Table 2.14.

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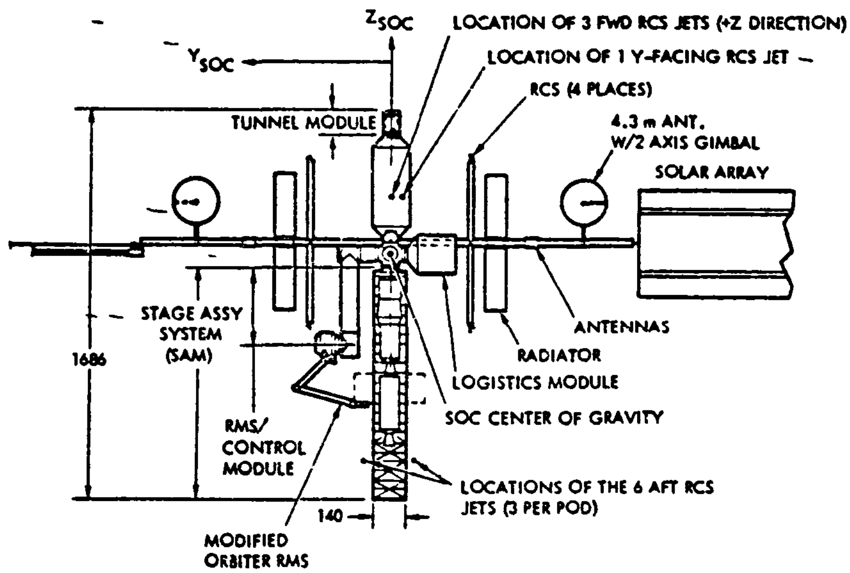

FIGURE 2.39 END VIEW OF THE SOC CONFIGURATION USED FOR PLUME IMPINGEMENT ANALYSIS (LOOKING FROM THE ORBITER)

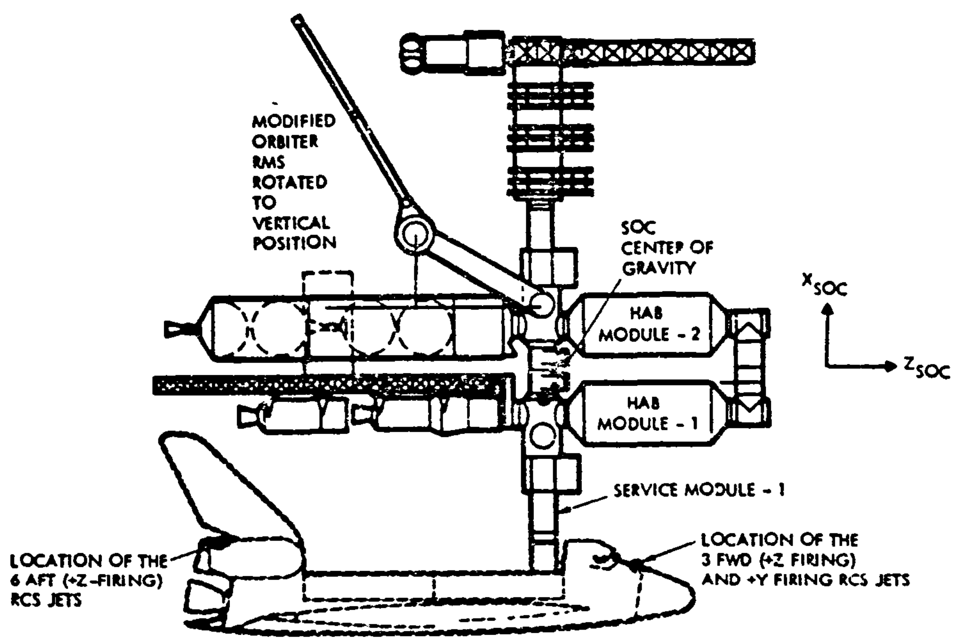

FIGURE 2.40 SIDE VIEW OF THE SOC/ORBITER CONFIGURATION USED FOR PLUME IMPINGEMENT ANALYSIS

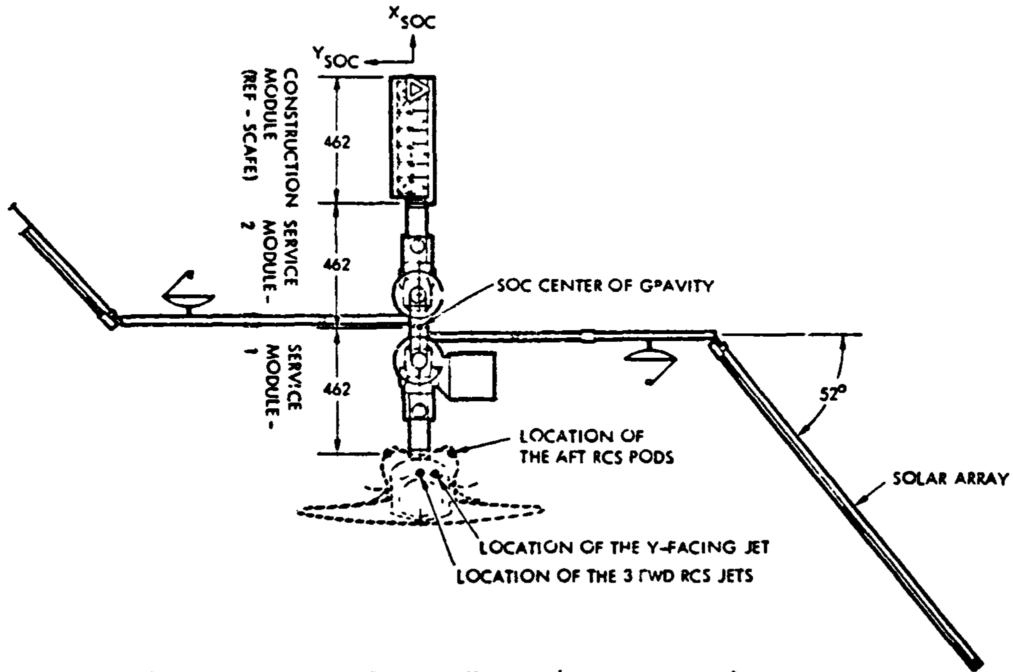

FIGURE 2.41 TOP VIEW OF THE SOC/ORBITER CONFIGURATION FOR PLUME IMPINGEMENT ANALYSIS

TABLE 2.14 SOC ELEMENTS AFFECTED BY PLUME IMPINGEMENT

FORWARD +Z RCS THRUSTERS (GROUP 4, THREE ENGINES)

- HABITABILITY MODULE NO. 1 (NEAREST DOCKED ORBITER)
- SERVICE MODULE NO. 1 (TO WHICH THE ORBITER IS DOCKED)
- LOGISTICS MODULE
- SOLAR ARRAY/SOC RCS SUPPORT BOOMS
- 4.3-M ANTENNAS AND ASSOCIATED BOOMS
- RADIATORS
- R/CM UPPER STRUCTURE

AFT +Z RCS THRUSTERS (GROUPS 11 AND 12, SIX ENGINES)

- VEHICLES PARKED FOR SERVICING (e.g., PLANETARY STAGES)
- STAGE ASSEMBLY MODULE (SAM)
- R/CM LOWER STRUCTURE AND CAB
- LOWER RCS BOOMS
- LOWER RADIATOR PANELS
- LOGISTICS MODULE

FORWARD -Y THRUSTER (GROUP 3, ONE ENGINE)

- SOLAR ARRAY AT 52-DEGREE ANGLE FROM BOOM (MAXIMUM SURFACE)
- -Y 4 3-M ANTENNA AND BOOM
- -Y RADIATORS
- -Y RCS/SOLAR ARRAY BOOMS
- LOGISTICS MODULE/HABITABILITY MODULE NO 1

To permit evaluation of the local heating and plume pressure environments on these various SOC elements basic plume models were established; first for the individual thrusters and then for the composite environments associated with the nine Z-jet abort thrusting and two Y-thrusters associated with deviation correction maneuvers. The plume pressure and heating characteristics for an individual thruster are shown in Figures 2.42 through 2.45 for the forward and aft thruster locations and in Figures 2.46 through 2.49 for the two multi-thruster environments.

Various local plume conditions anywhere on the SOC configuration can be estimated using these basic plume data. The effects of different combinations of jet firings and orbiter orientation geometries can also be determined by adding the individual thruster plume characteristics from Figures 2.42 through 2.45, accounting for the appropriate plume direction and distance factors for the cases to be evaluated. Conditions for future SOC configurations and/or new facility features can also be approximated in this way.

To assess the potential magnitude of plume impacts on SOC operations the main disturbance effects and heating environments were determined for the baseline SOC configuration (Figures 2.39, 2.40 and 2.41). Emphasis was on the moment producing force elements, thus certain symmetrical impingement effects were ignored. The resulting main impingement influences from the forward thrusters are on Habitability Module No. 1, the Logistics Module and the Service Module end facing the orbiter; for the aft thrusters the main affects are on the planetary stages (or any vehicle parked on the SAM for servicing), the stage assembly module and the RCM module; and for the Y-thrusters the principal effect is mostly on the solar array with small influences on the 4.3 meter antenna and radiator panels.

The effects of the Z-thruster (forward and aft) on the solar array and boom mounted equipment are small compared to their effects on the main items listed above. The same is true for the Y-thruster effects on the central cluster of modules which are small compared to the Z-thruster effects on these modules. The overall results are summarized in Table 2.15. These data reflect the basic pressure and heating contours in Figures 2.42 through 2.45, but also account for the local angle of incidence of the flow streamlines. To simplify the calculations for the small surface areas (antenna, radiators, etc.) the flow conditions at the point on the SOC configuration element represented by the centroid of the exposed area were assumed to apply over the entire surface of that element. The larger configuration elements were divided into several area segments with the flow conditions averaged over each segment.

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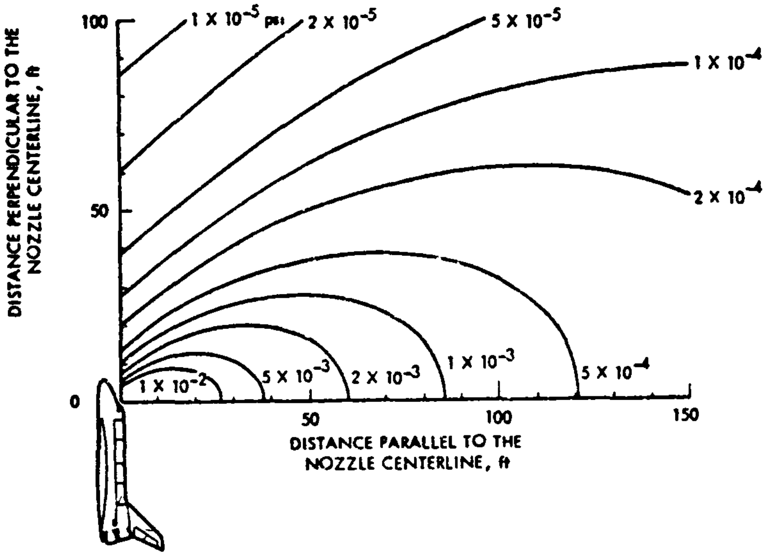

FIGURE 2.42 ISO-PRESSURE MAP FOR ONE FORWARD +Z FIRING RCS THRUSTER

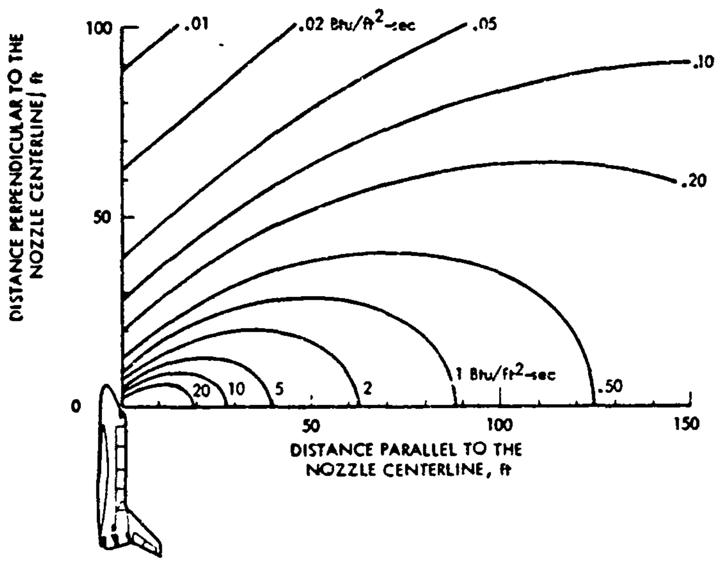

FIGURE 2.43 ISO-HEATING MAP FOR ONE FORWARD +Z FIRING RCS THRUSTER

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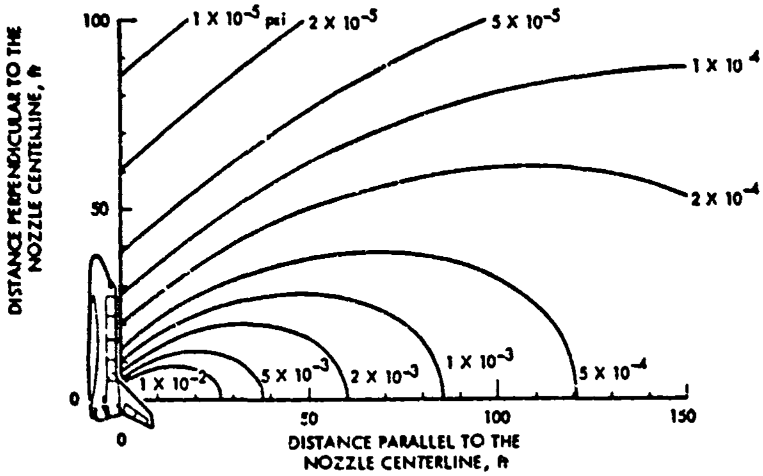

FIGURE 2.44 ISO-PRESSURE MAP FOR ONE AFT +Z FIRING RCS THRUSTER

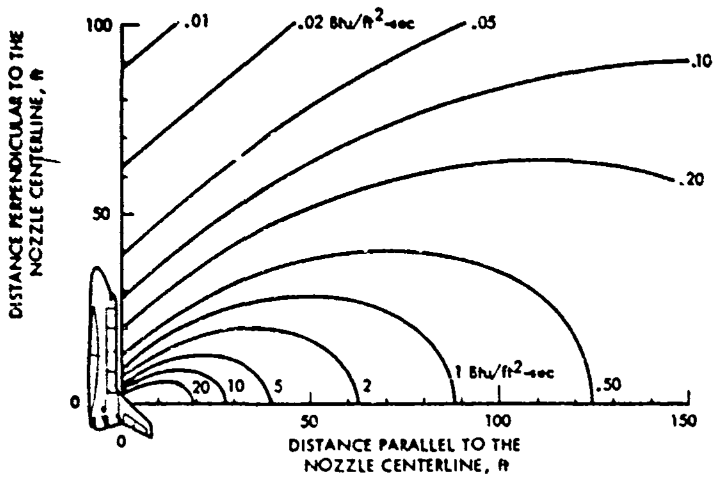

FIGURE 2.45 ISO-HEATING MAP FOR ONE AFT +Z FIRING RCS THRUSTER

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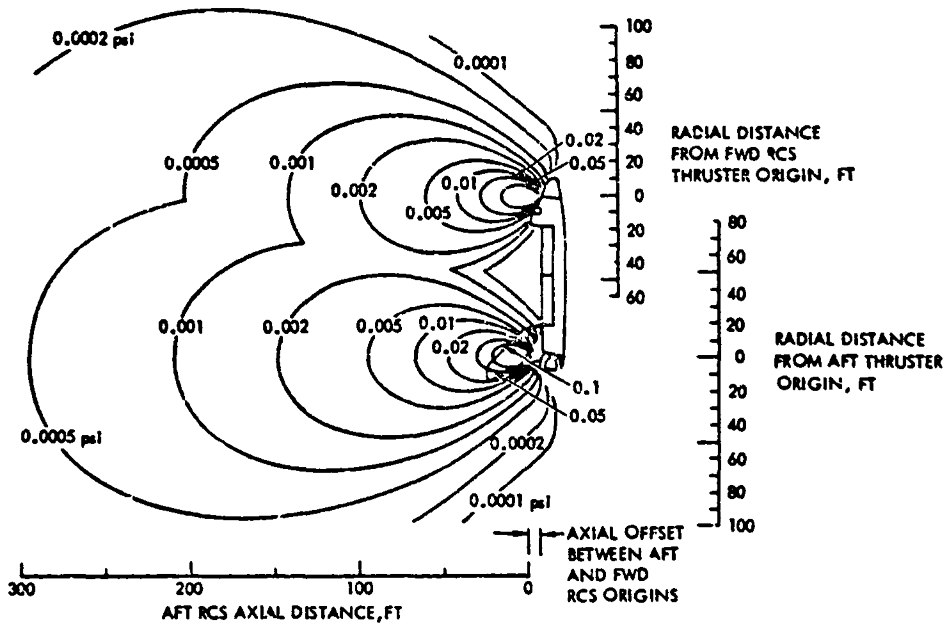

FIGURE 2.46 PLUME PRESSURE CONTOURS FOR 9 Z-THRUSTERS FIRING

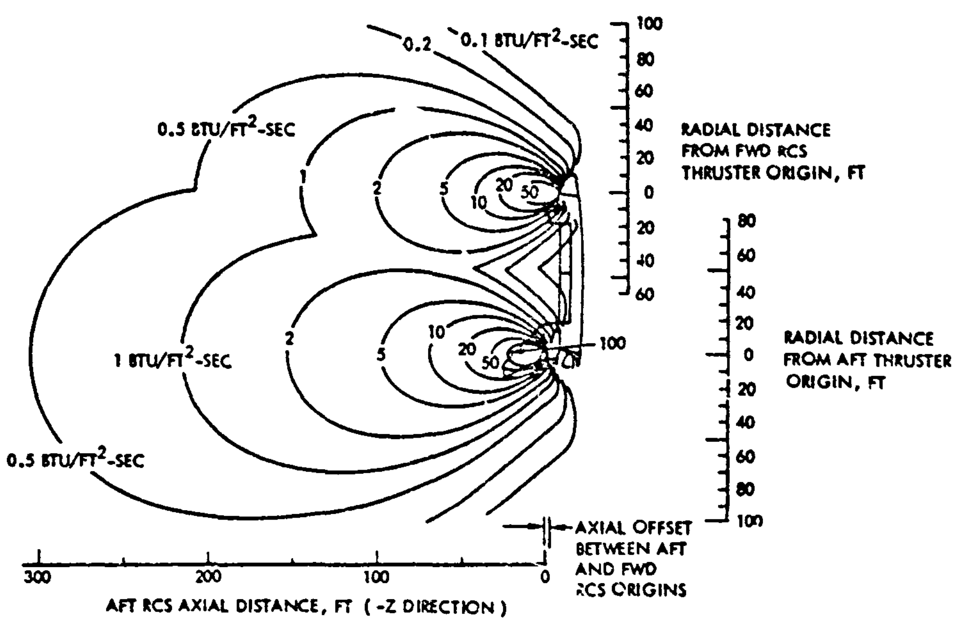

FIGURE 2.47 PLUME HEATING RATE FOR 9 Z-THRUSTERS FIRING

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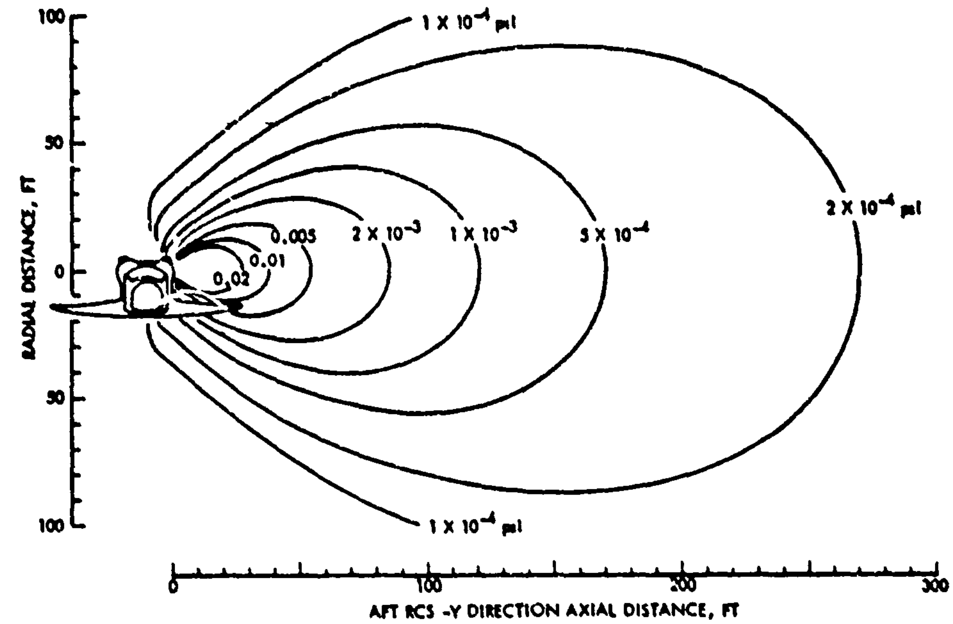

FIGURE 2.48 PLUME PRESSURE CONTOURS FOR 2 Y-THRUSTERS FIRING

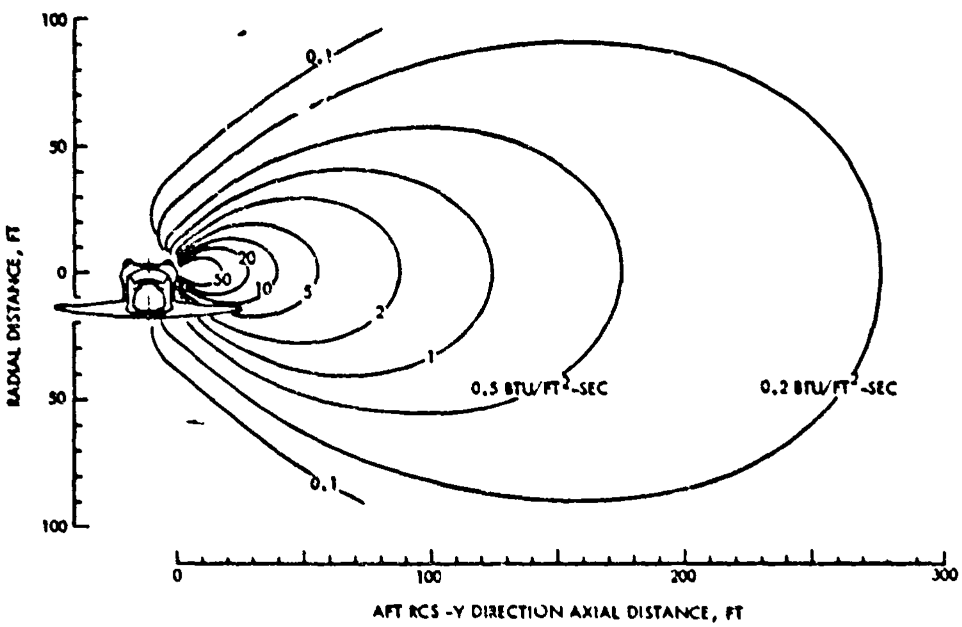

FIGURE 2.49 PLUME HEATING RATE CONTOURS FOR 2 Y-THRUSTERS FIRING

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TABLE 2.15 ORBITER RCS PLUME IMPINGEMENT RESULTS ON SOC

DESCRIPTION	MASS* DEPOSITION RATE (lbm/sec)	SOC IMPINGEMENT FORCES (lb_f)			SOC MOMENTS (lb_f-ft)			CONVECTIVE HEATING RATE (Btu/sec)
		F_X	F_Y	F_Z	M_X	M_Y	M_Z	
FWD RCS, 3 ENGINES (+Z DIRECTION)								
HABITABILITY MODULE NO. 1	3.146	981.1	0	41.8	0	24,058.1	0	7730.0
LOGISTICS MODULE	0.272	59.7	-11.7	-60.6	809.0	-508.2	699.0	637.7
SERVICE MODULE NO. 1	0.266	41.2	0	-29.0	0	-189.4	0	603.5
TOTAL	3.684	1082.0	-11.7	-47.8	809.0	23,360.5	699.0	
AFT RCS, 6 ENGINES (+Z DIRECTION)								
PARKED PLANETARY VEHICLE	0.154	90.6	0	50.8	0	-5,185.7	0	773.2
SAM**	2.566	867.2	0	-70.2	0	-67,579.0	0	6540.6
A/CM MODULE	0.280	60.0	23.5	39.2	1758.7	-3,255.6	-247.9	569.8
TOTAL	3.198	1017.8	23.5	19.8	1758.7	-76,020.3	-247.9	
-Y THRUSTER, 1 ENGINE								
SOLAR ARRAY (@ 52° ANGLE)	0.720	116.7	-171.4	-45.1	6018.4	-1957.3	19,554.1	1659.8
6.5 M ANTENNA (-Y DIRECTION)	0.036	3.2	-5.4	-0.4	123.2	55.2	220.6	45.9
RADIATORS (-Y DIRECTION)	0.009	2.3	-1.4	-0.5	35.1	20.3	29.0	20.6
TOTAL	0.765	122.2	-178.2	-46.0	6176.7	-1881.8	19,853.8	

*NOTES: (1) ONE ENGINE PRODUCES 870 lb_f THRUST
(2) MASS FLOW RATE OF ONE ENGINE—3.01 lbm/sec
(3) MASS FLUX CONTAINS APPROX 5% CO_2, 17.5% CO, and 29.2% H_2O

**ASSUMED THAT THE SAM WAS OPAQUE (INTERNAL PARTS STOWAGE)

The results show there could be considerable mass deposition and impingement heating on the Habitability Module from the forward RCS thrusters. In addition, relatively high system torques can be induced by impingement forces, particularly pitch up moments. The "pitch up" direction of these induced moments is caused by the orbiter RCS location geometry in the near docking orientation. The flow field center of the nine-thruster plume is somewhat below the docking port and hence, more heavily influences the SOC configuration elements mounted beneath the service modules. This in turn causes the general pitch up tendency. Even greater pitch up torques could be induced if large OTV stages and crew modules were parked on the SAM or other flight support facility installations beneath the Service Modules.

In addition to the disturbance torque effects on the SOC control system, OTVs parked in this region would be subjected to significant plume bombardment on their lightweight thermal insulation blankets. Vacuum tests with 0.1 lbf hydrazine thrusters impinging on 2 mil thick aluminized MLI have shown the shearing forces to cause more damage than heating effects. Thus, shielding protection from the plume may be required. This may also be a problem with the SOC RCS thrusters as well although they are smaller and less directly aimed. Orbiter abort thrusting with the nine Z-thrusters will be relatively brief, probably about two seconds maximum, but potential MLI damage may result and needs further, more detailed study.

The Y-thrusters can cause significant yaw torques from impingement on the solar array. In addition, mass deposition rates of 0.7 lbm/sec (24 percent of the available mass) are possible on the solar array. These are for the worst case geometry (sun β-angles of 52 deg) which would not exist all of the time. Planning SOC resupply missions to avoid these worst case conditions is possible. A few days' delay or advancement of a resupply mission could significantly reduce the peak sun angle effects. Plume impingement effects on the solar arrays could also be further reduced by "feathering" procedures to align the surfaces with the RCS plume field. This would interrupt normal power generation (if performed on the daylight side of the orbit) and lead to more complex SOC operations, but could greatly reduce SOC design problems if warranted.

2.3 DOCKING/BERTHING DESIGN CONCEPT

The design concept effort consisted of two principal tasks; (1) define a standard docking/berthing interface configuration, and (2) develop an orbiter docking module concept.

2.3.1 Docking/Berthing Interface Concept

Standard Docking/Berthing Interface Utilities Arrangement

The definition of standard refers not only to the interface ports for the SOC modules and the orbiter docking module, but also to attach ports of other spacecraft that would want to berth/dock to the orbiter or to the SOC. Table 2.16 lists some of the significant requirements that were identified for the docking/berthing system. In addition to these physical requirements, the utilities to be connected across the mated interface needed to be defined. Final number and arrangement of the utilities is dependant on the integration of the SOC individual module requirements and the utilities requirements of other candidate spacecraft. However, in order to define a standard interface at this time, when these SOC requirements are unknown, the utilities defined from previous space station studies were utilized as a model and modified to be more compatible with the SOC concept. Figure 2.50 lists the interface utilities anticipated to be connected across the port and also illustrates the arrangement within the berthing port.

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TABLE 2.16 DOCKING/BERTHING INTERFACE REQUIREMENTS

- THE DOCKING/BERTHING SYSTEMS SHALL BE ANDROGYNOUS, EXCEPT FOR CERTAIN SPECIALIZED UMBILICAL INTERCONNECTS (i.e., MODULES WITH IDENTICAL DOCKING SYSTEMS MAY BE DOCKED TOGETHER, AND MODULES WITH IDENTICAL BERTHING SYSTEMS MAY BE BERTHED TOGETHER.
- ALL BERTHING PORTS ON THE SOC BASIC CONFIGURATION SHALL BE GEOMETRICALLY IDENTICAL, EXCEPT FOR SPECIALIZED UMBILICAL INTERCONNECTS, SUCH THAT ANY MODULE OF THE SOC MAY BE BERTHED TO ANY BERTHING PORT—OTHER CONFIGURATION FACTORS PERMITTING.
- THE SOC BERTHING PORTS AND THE ORBITER'S DOCKING SYSTEM, LOCATED ON ITS DOCKING MODULE, SHALL BE DESIGNED SUCH THAT THE ORBITER MAY DOCK TO ANY SOC BERTHING PORT.
- THE BERTHING AND DOCKING SYSTEMS SHALL BE DESIGNED TO ALLOW DOCKING AT 90-DEGREE ALIGNMENT INCREMENTS.
- ALL BERTHING/DOCKING SYSTEM HARDWARE SHALL BE LOCATED INSIDE THE PRESSURE-SEALED INTERFACE, AND BE ACCESSIBLE TO THE CREW FROM WITHIN THE PRESSURIZED ENVIRONMENT.
- ALL UMBILICAL INTERCONNECTS SHALL BE LOCATED INSIDE THE PRESSURIZED ENVIRONMENTS.
- ALL ACTIVE FUNCTIONS OF THE BERTHING/DOCKING SYSTEM (i.e., IMPACT ATTENUATION, CAPTURE LATCHING, STRUCTURAL LATCHING, ETC.) SHALL NORMALLY BE PERFORMED BY ONE SIDE, WITH THE OTHER SIDE IN A PASSIVE MODE.
- A ONE-METER-DIAMETER CLEAR PASSAGEWAY SHALL BE PROVIDED WITHOUT REMOVAL OF MECHANISMS.
- DOCKING DESIGN IMPACT CONDITIONS

- AXIAL CLOSING VELOCITY	0.48-0.15 m/sec (0.16-0.50 ft/sec)
- LATERAL VELOCITY	<0.06 m/sec (0.2 ft/sec)
- ANGULAR VELOCITY	<0.6 deg/sec
- LATERAL MISALIGNMENT	0.23 m (0.75 ft)
- ANGULAR MISALIGNMENT	<5.0 deg, ROLL; <6.0 deg, PITCH/YAW

The utility connections are made only after the structural mating has been made and verified. The utility connections are remotely actuated for both electrical and hard lines, fluid and gas transfer. Figure 2.51 illustrates the remote actuator concept(s) that is compatible with the arrangement shown in Figure 2.50. The top surface of the connectors are located approximately 1 cm below the mating interface. This a rangement will prevent damage to the connectors in the event of an off nominal docking/berthing maneuver. Manual override capabilities are provided to accommodate anomalies. Shirtsleeve or suited maintenance activities can be accommodated (Figure 2.52).

Not all the utilities illustrated in Figure 2.50 would be required to interconnect across each interface. However, the location of those utilities that are required would be at those dedicated positions. This arrangement will permit all spacecraft docking/berthing to the orbiter to be compatible in a standard pattern. Figure 2.53 shows an example of a SOC module to module berthing arrangement, and an orbiter docking module to SOC or other spacecraft docking/berthing arrangement.

Standard Docking/Berthing Interface Configuration

Both the passive and the active port configurations are illustrated in Figure 2.54 Drawing 42690-020 in Appendix A defines these ports in more detail.

The size of this interface is governed by the 1.0 M clear opening and the space required for the utilities, latches and pressure seal. All of the utilities interface are within the pressurized area with the pressure seal, therefore, located at the extreme outboard edge of the interface surface. Eight structural latches located among the utilities provide the final physical connection between the mating surfaces. Four of the active latches are provided on one side and four redundant latch strikers are located on the other side. Four externally oriented guides provide the alignment for the mating operation. Each guide contains a capture latch which secures the vehicles until the final structural latch is made.

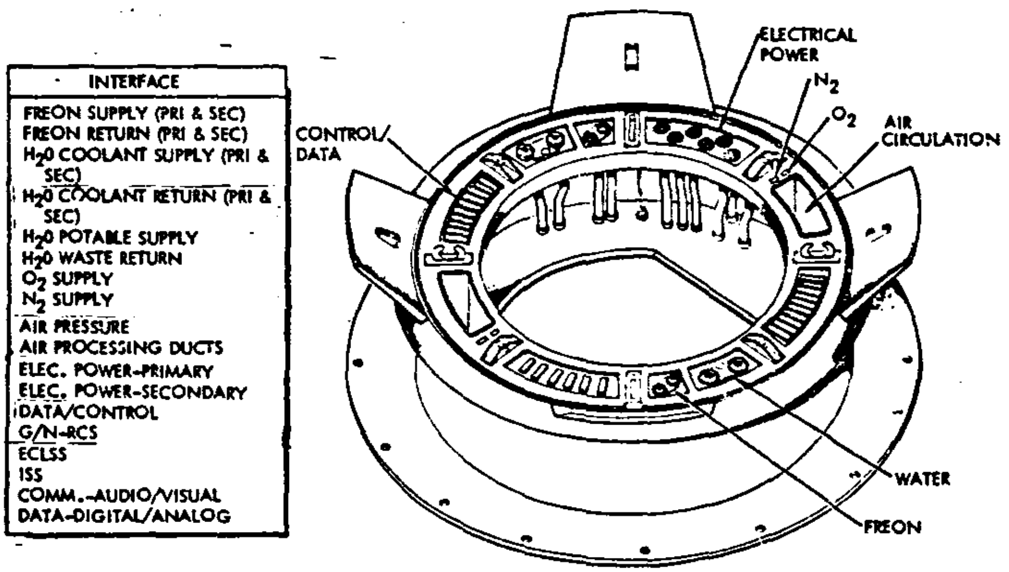

FIGURE 2.50 UTILITIES INTERFACES ARRANGEMENT

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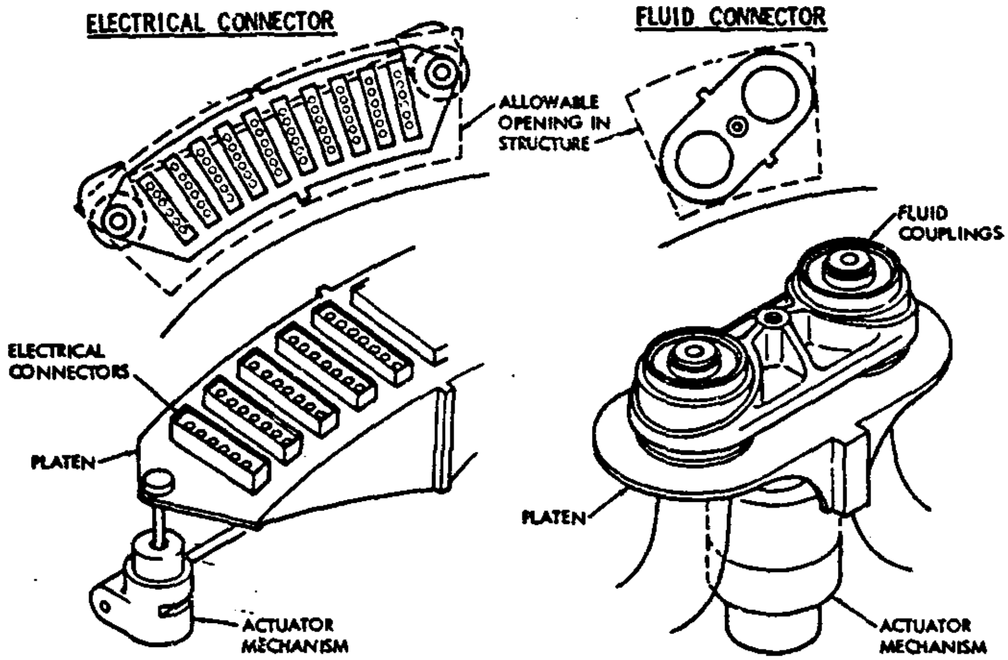

FIGURE 2.51 REMOTE ACTUATED INTERFACE CONNECTOR CONCEPTS

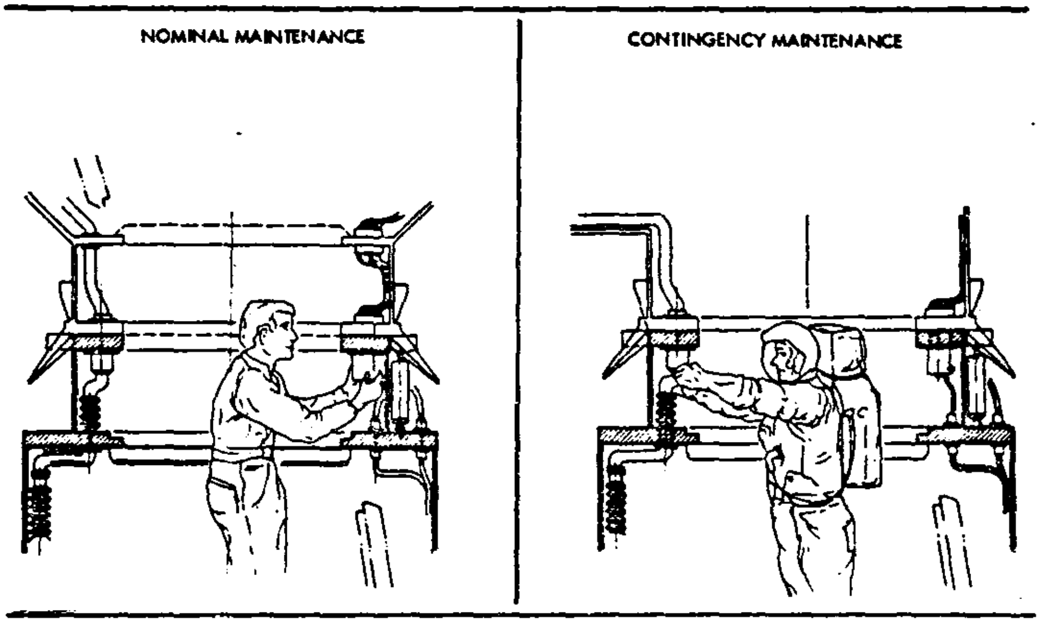

FIGURE 2.52 UTILITIES INTERFACE ACCESS

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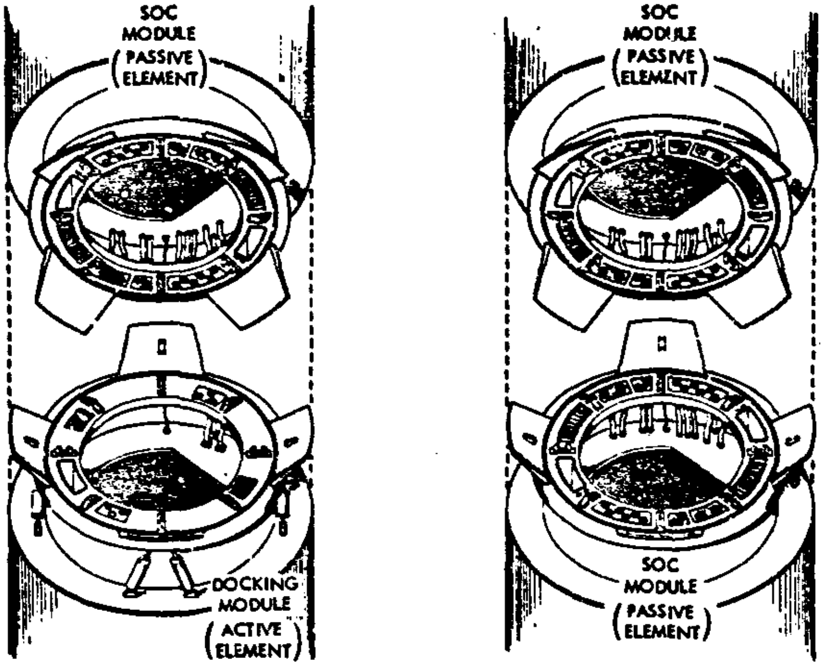

FIGURE 2.53 MATING ARRANGEMENTS

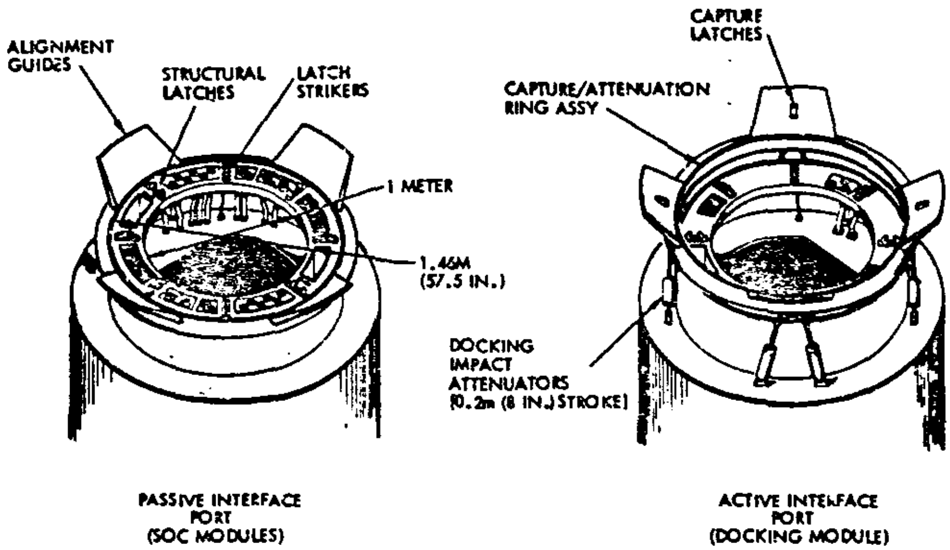

FIGURE 2.54 STANDARD DOCKING/BERTHING PORT CONFIGURATION

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A passive port can accommodate a berthing mating operation. No active attenuation is anticipated other than the inherent structural compliance. However, an attenuation system is necessary to accommodate docking operations with impact velocities of up to .15 M/sec (0.5 ft/sec). A .2 M (8 inch) stroke will accommodate this impact velocity with reasonable attenuation. An investigation was made to verify that the docking module utilizing an attenuation stroke of .2 M would always provide clearance between the orbiter and the mating elements. A 6° misalignment was used as a maximum mismatch between the docking module and the mating element. The worst possible attitude is shown in Figure 2.55, deflecting the misalignment to the maximum angle of 7°. With the 6° misalignment added, the module is pitched toward the cabin roof at an angle of 13°. At this position, there is a clearance of five inches between the maximum diameter of the module and the TPS tiles. This condition conservatively assumed that the attenuators immediately "bottomed out" and did not impose any restoring moments. The investigation also assumed that the orbiter was a fixed element which did not rotate and thus relieve the impact force. A dynamic analysis of this condition that considers these elements is necessary to provide a final answer to the question. However, this worst-worst case investigation indicated that the docking module interface location is acceptable with a 6° misalignment for safe docking operations.

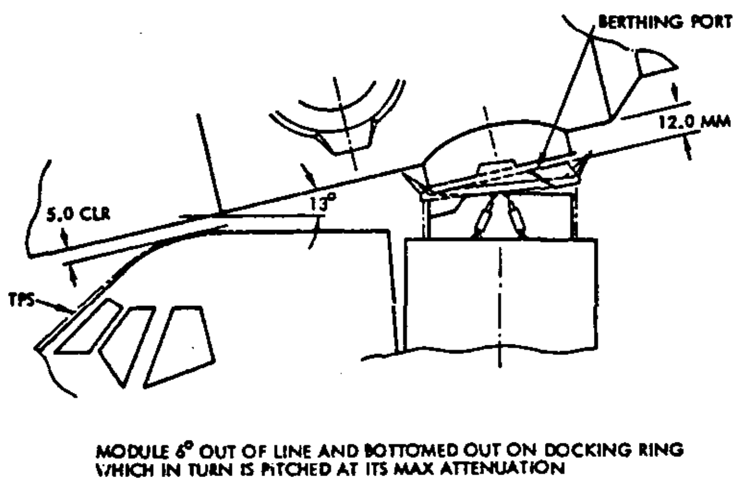

FIGURE 2.55 CLEARANCES WITH MISALIGNED DOCKING

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This active port has the guide pedals mounted to a ring which is supported by four pairs of attenuation struts. Contact between the mating vehicles occurs at the ring and the guide pedal capture latches. A ring retract mechanism will pull the mating surfaces together to affect the structural latching and pressure sealing.

The attenuators are located externally to the pressurized volume. This arrangement permits clear access to the utilities interface mechanisms and provides support to the attenuation ring at the most desirable location. The arrangement also permits the option of utilizing hydraulic attenuators which would not be permitted within the pressurized volume.

Other Spacecraft Mating Concepts

In order to define a standard mating interface concept, other spacecraft projects that are under study were investigated to understand their requirements and mating concepts. Four projects as well as the Shuttle Orbiter baseline docking system were analyzed. The four projects are illustrated in Figure 2.56; the 25 KW Power System, the Science and Applications Space Platform (SASP), space construction of the Engineering Technology Verification Platform (ETVP), and the Flight Support System (FSS) of the Multi Mission Spacecraft (MMS) System. The mating mechanisms and utilities interface concepts of each of the projects Figures 2.57 and 2.58, were evaluated and compared to the SOC developed concept. In all cases the SOC concept could be substituted for the mating concepts developed for each project with the possible exception of the FSS which has hardware fabricated at this time. A typical installation is illustrated in Figure 2.59 which shows the installation with a 25 KW Power System.

2.3.2 Orbiter Docking Module Concept

The principal function of the docking module is to provide a pressurized, shirtsleeve, passageway between the orbiter crew cabin and the SOC. The significant requirements imposed on the design of the docking module concept are listed in Table 2.17. Figure 2.60 illustrates the concept developed that complies with these requirements.

Docking Module Configuration

Two docking module configuration options were identified and analyzed (Figure 2.61). One concept consisted of a single unit that interfaced to the aft bulkhead of the crew cabin and was supported from the payload support bridge fittings. This unit also incorporated the Spacelab tunnel interface at 660. A flexible joint at the aft crew cabin bulkhead isolates the docking loads from the cabin.

The second option utilizes the Spacelab tunnel adapter as the lower portion of the docking module. The interface to the orbiter cabin is, therefore, at the tunnel adapter. This interface is the same interface that is utilized for an external airlock payload bay arrangement. The docking loads are reacted via the payload bridge fittings with a flexible, load

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isolation joint at the tunnel adapter interface. The tunnel adapter provides the interface to the Spacelab tunnel at 660. This concept provides the capability to install or remove the orbiter mating capability in the same manner as the removal or installation of a payload. The concept greatly minimizes the turnaround activity required for this type of reconfiguration. Figure 2.62 illustrates this flexibility. This option was, therefore, selected as the docking module concept.

The electrical power and the data/control interfaces to the orbiter utilize the dedicated payload feed through panels in the cabin aft bulkhead for the data/control interface, and interface for the electrical power at the wire tray on either side of the payload bay. This arrangement is indicated in Figure 2.62 and 2.63. These interfaces accommodate the power and control necessary for the operation of the docking module as well as the requirements across the mating interface. Control of the electrical lines, both power and data, during extension and retraction of the docking module is provided by coiling the wires, approximately 4" dia coil, around a central telescoping guide rod. Fluid and gas distribution lines accommodate the extension/retraction motion with flexible, bellows type, tubes, indicated in Figure 2.63.

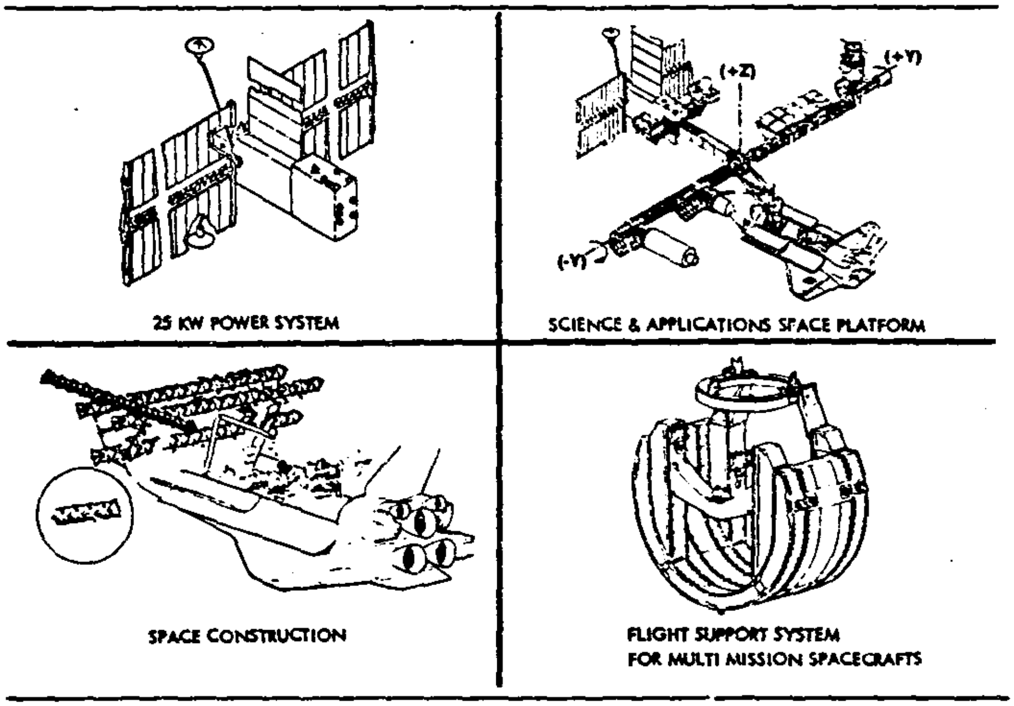


FIGURE 2.56 SPACECRAFT PROJECTS THAT MATE WITH THE ORBITER

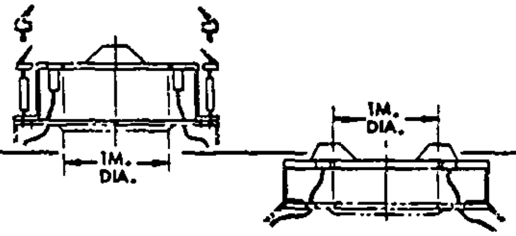
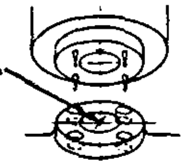
PROJECT	REQUIREMENTS	CONCEPT
ORBITER (MCR 5546)	• CONTACT VELOCITY -Z - 0.5 FPS -0.5 FPS X, Y -0 ± 0.1 FPS ANGULAR - ± 1.0° SEC • MISALIGNMENT LATERAL -0 ± 0.5 FT ANGULAR -0 ± 5 DEG ROTATIONAL -0 ± 7 DEG	
SOC/ORBITER	• CONTACT VELOCITY -Z -0.16 FPS - 0.5 FPS X, Y - 0.2 FPS ANGULAR - 6°/SEC • MISALIGNMENT LATERAL - ± 0.75 FT ANGULAR - ±6 DEG ROTATIONAL - ±5 DEG	
SOC MODULE/ SOC MODULE	• CONTACT VELOCITY TBD • MISALIGNMENT LATERAL - ±2 IN. ANGULAR - ± 1 DEG	
25 KW POWER SYSTEM	• CONTACT VELOCITY TBD • MISALIGNMENT LATERAL - ±2 IN. ANGULAR - ±1 DEG	

FIGURE 2.57 MATING MECHANISMS (1)

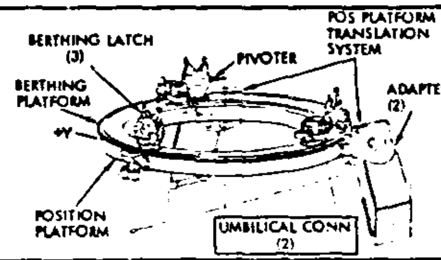
PROJECT	REQUIREMENTS	CONCEPT
SPACE PLATFORM	• CONTACT VELOCITY -Z -0.1 FPS X, Y -0.1 FPS ANGULAR - 1°/SEC • MISALIGNMENT LATERAL - ± 0.33 FT ANGULAR - 10° ROTATIONAL - 10°	
SPACE CONSTRUCTION	• CONTACT VELOCITY TBD • MISALIGNMENT LATERAL - ±2 IN. ANGULAR - ±1 DEG	
MMS/FSS	• CONTACT VELOCITY 0.5 FPS • MISALIGNMENT LATERAL - ±2 IN. ANGULAR - ±1 DEG	

FIGURE 2.57 MATING MECHANISMS (2)

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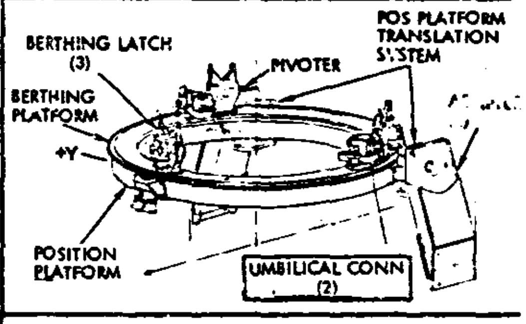
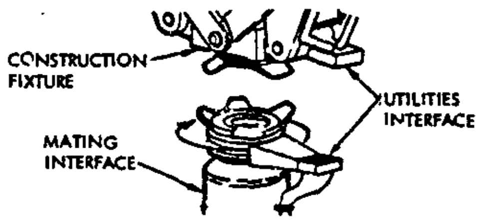
PROJECT	REQUIREMENTS	CONCEPT
SPACE PLATFORM	SAME AS 25 KW POWER SYSTEM	
MMS/FSS	• REMOTE ACTIVATION • MANUAL BACK-UP SEPARATION	
SPACE CONSTRUCTION	• REMOTE ACTIVATION	

FIGURE 2.58 UTILITIES INTERFACES (1)

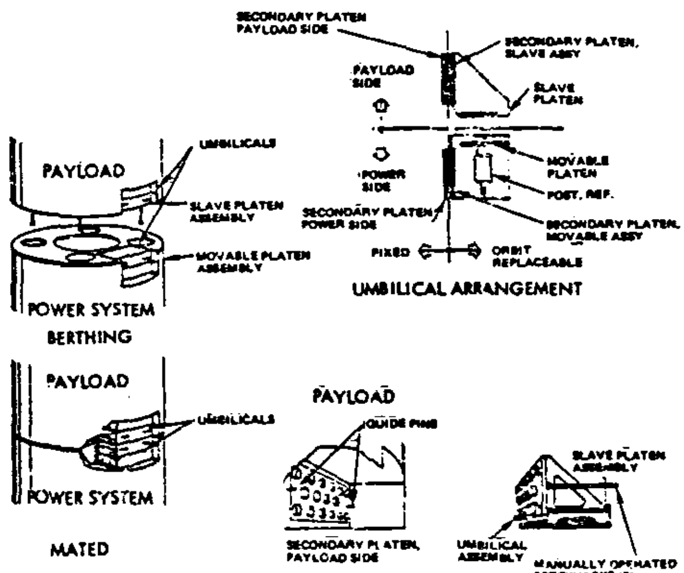
PROJECT	REQUIREMENTS	CONCEPT
25 KW POWER SYSTEM	• EVA ACCESS • REMOTE ACTIVATION • ON ORBIT REPLACEABLE UMBILICAL	

FIGURE 2.58 UTILITIES INTERFACES (2)

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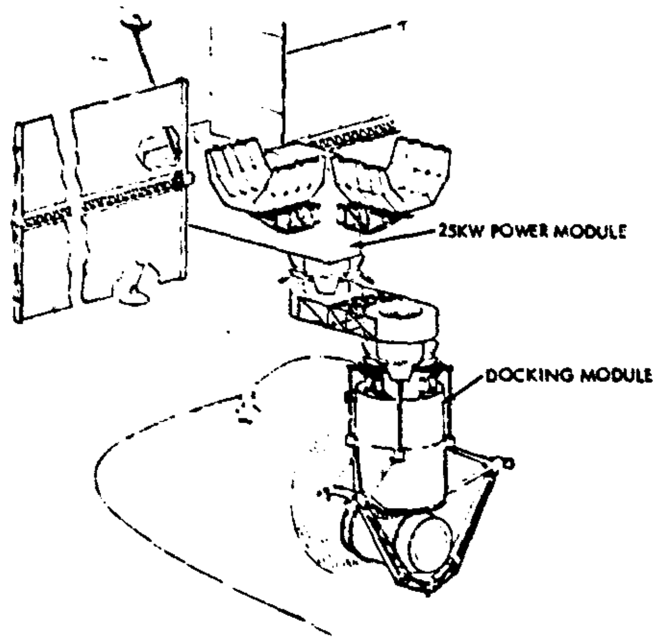

FIGURE 2.59 25KW POWER SYSTEM WITH DOCKING/BERTHING CONCEPT

TABLE 2.17 ORBITER DOCKING MODULE REQUIREMENTS

- THE DOCKING MODULE WILL PROVIDE A PRESSURIZED PASSAGEWAY BETWEEN THE ORBITER CREW CABIN AND THE BERTHING/DOCKING INTERFACE.
- A ONE-METER CLEAR PASSAGE WILL BE PROVIDED.
- A SPACELAB TUNNEL INTERFACE WILL BE PROVIDED AT $X_0$660, $Z_0$357.9.
- IN THE RETRACTED POSITION, A 36-IN. CLEAR PASSAGE WILL BE MAINTAINED BETWEEN THE INNER MOLD LINE OF THE CARGO BAY DOORS AND THE MATING INTERFACE OF THE DOCKING MODULE.
- THE MATING INTERFACE WILL EXTEND TO $Z_0$515 AS A MINIMUM.
- AN EMERGENCY SEPARATION WILL BE PROVIDED SUCH THAT SUBSEQUENT PAYLOAD BAY DOOR CLOSURE IS POSSIBLE AND THAT THE SEPARATION WILL NOT RESULT IN A LOSS OF CABIN PRESSURE, OR CREATE UNACCEPTABLE VEHICLE DAMAGE OR LOSS OF EVA CAPABILITY.
- THE DOCKING MODULE CONCEPT WILL NOT PRECLUDE THE CAPABILITY TO FUNCTION AS AN AIRLOCK.
- THE DOCKING MODULE INSTALLATION WILL ACCOMMODATE THE SERVICING AND OPERATION OF ONE MMU.

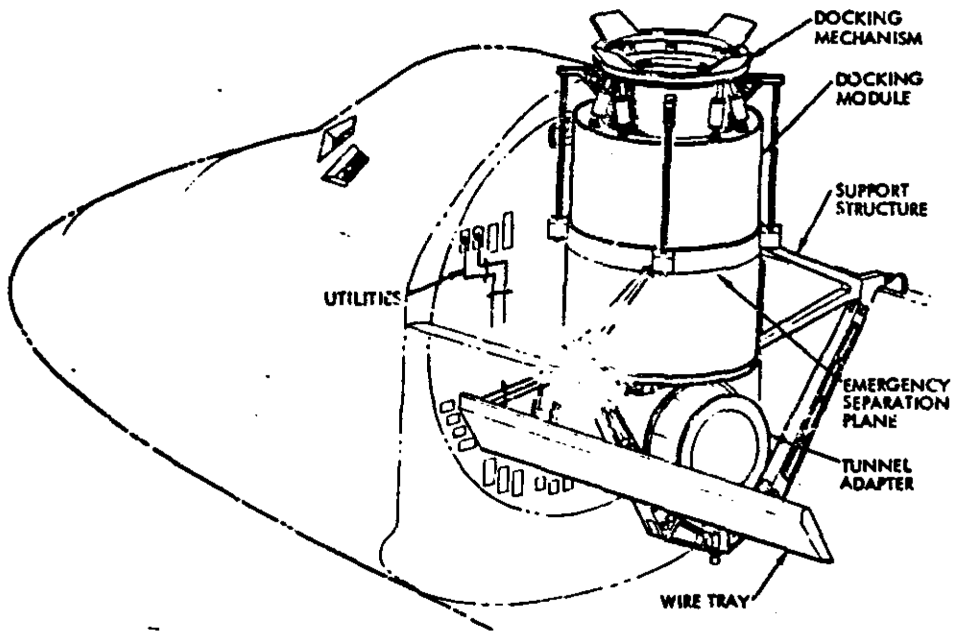

FIGURE 2.60 DOCKING MODULE CONCEPT

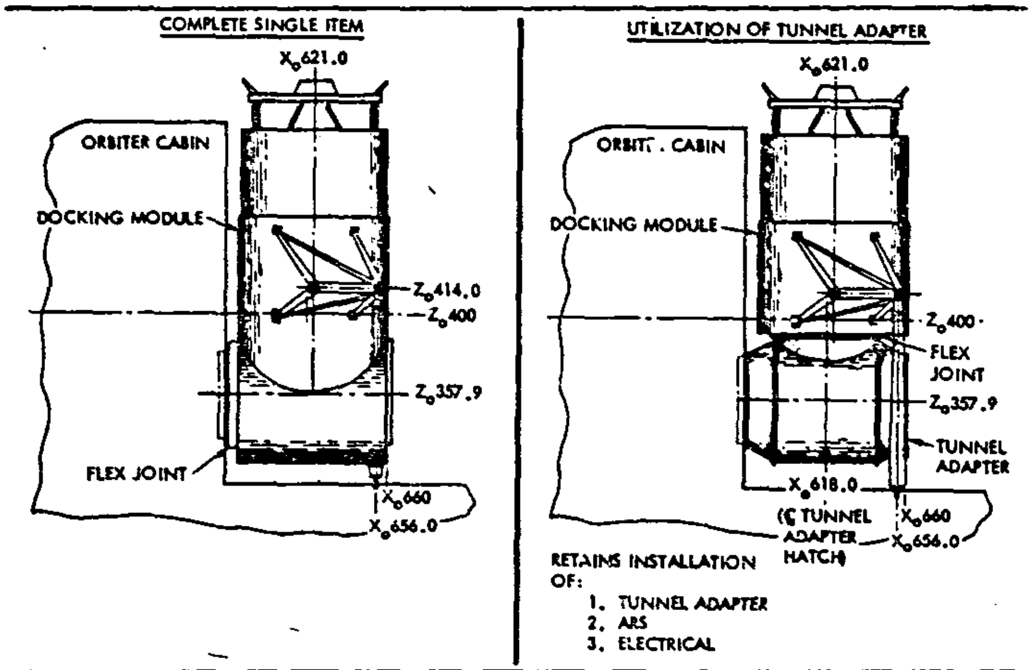

FIGURE 2.61 DOCKING MODULE CONFIGURATION OPTIONS

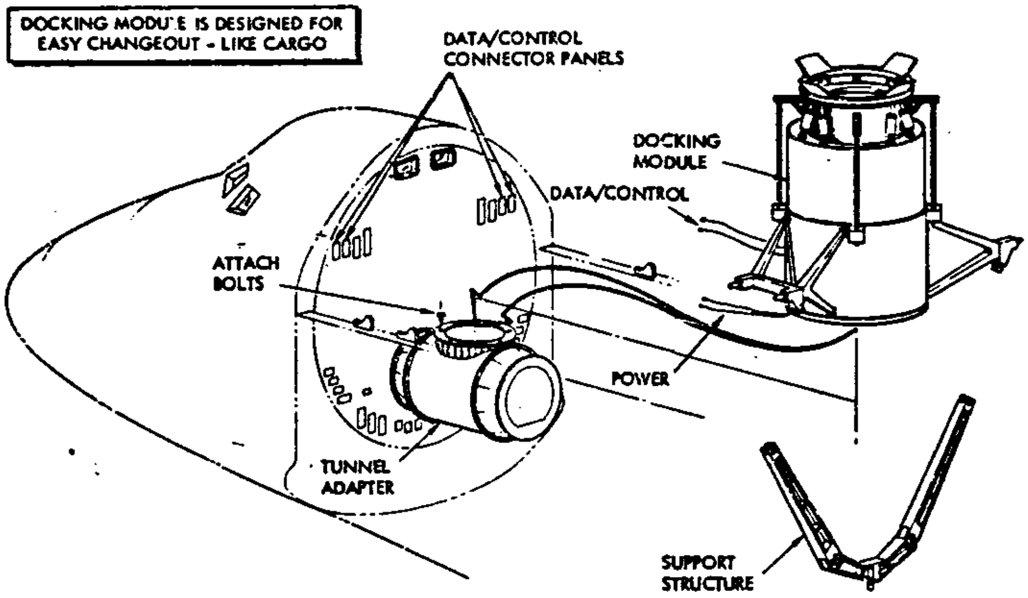

FIGURE 2.62 DOCKING MODULE INSTALLATION

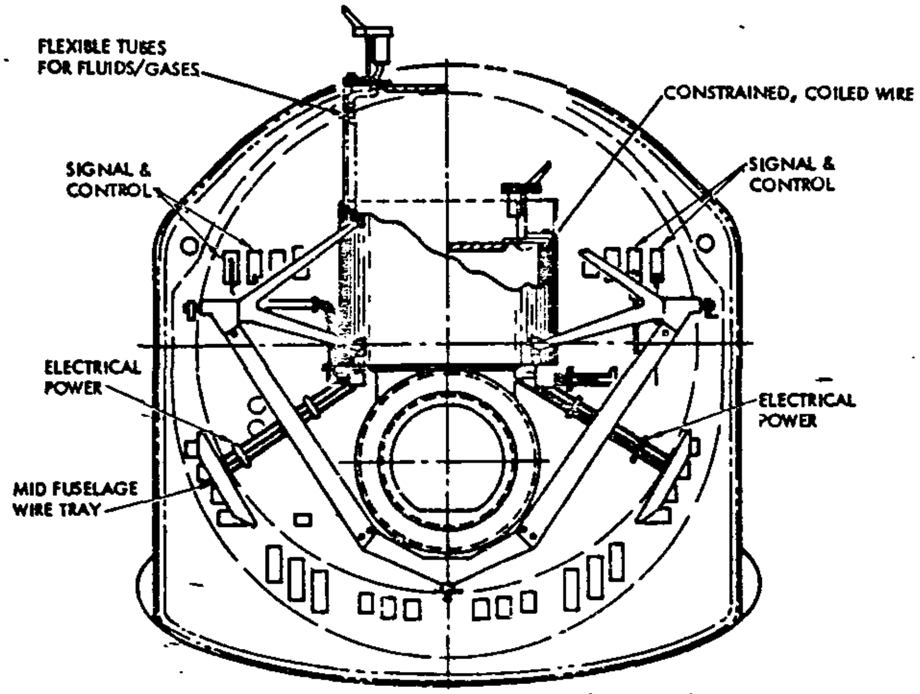

FIGURE 2.63 UTILITIES INTERFACE ROUTING

In addition to reacting the docking loads, the docking module support structure arrangement must provide clearance for the operation of one MMU and stowage provisions for the manned remote work station (MRWS). A structural arrangement to accommodate these requirements was developed and is indicated in Figure 2.64. Later information, however, resulting from the MRWS studies showed the inability of the orbiter remote manipulator system (RMS) to reach the stowage position of the folded MRWS. The stowage of the MRWS on the orbiter is moved aft, clear of the docking module support structure. Consequently, the final docking module support structure is as shown on Figure 2.65.

Docking Module EVA Operations

EVA capability must be maintained when the docking module is installed on the orbiter. EVA is accomplished by utilizing the on board airlock. The docking module, therefore, must provide a clear passage way to the ambient environment. Figure 2.66 illustrates the passage ways available for several programs. For three of the arrangements the on board airlock of the orbiter provides the EVA capability. However, if a Spacelab program requires simultaneous EVA and a clear Spacelab passage, then the docking module must become the airlock and this arrangement is indicated on the lower two examples. Airlock pressurization controls and an upper hatch are the only items required to have the docking module assume the role of an airlock. However, the Spacelab operation requiring simultaneous passage and EVA capabilities seems quite remote so that the nominal docking module arrangement selected does not have the airlock capability although provisions to accommodate this capability would be present. The nominal docking module concept is illustrated in Figure 2.67.

To perform EVA through the docking module for unmanned missions, such as the 25 KW Power System and SASP, clearance must be provided at the end of the docking module to permit EVA egress. Examples of this condition are illustrated in Figure 2.68. An alternative solution is illustrated in Figure 2.69 which indicates the feasibility of incorporating the standard docking/berthing interface on an unpressurized docking module concept. EVA egress can be accomplished by incorporating a clear passage through the truss structure of the unpressurized docking module design rather than perturbing the spacecraft.

Docking Module Requirements

Preliminary requirements have been generated to define the docking module. The requirements are grouped into the three principal design tasks; (1) the docking mechanism, (2) the utilities interfaces, and (3) the docking module primary structure and tunnel extension. These requirements are listed in Table 2.18.

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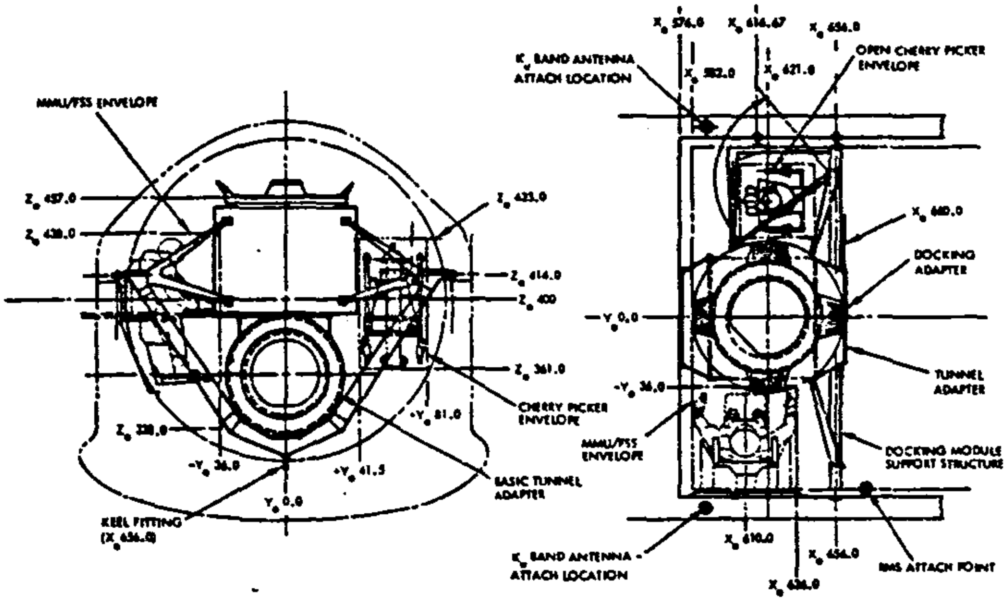

FIGURE 2.64 INITIAL DOCKING MODULE SUPPORT STRUCTURE

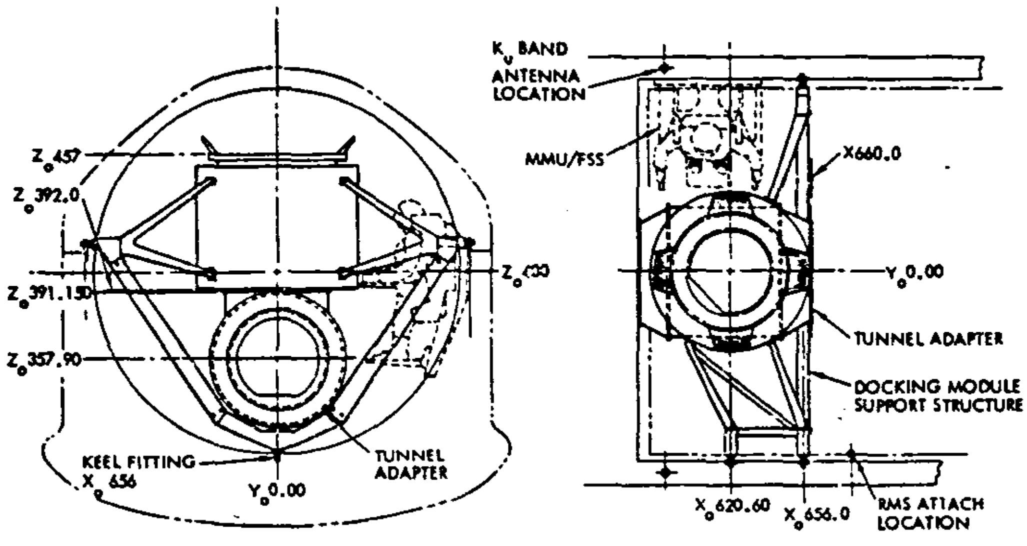

FIGURE 2.65 DOCKING MODULE SUPPORT STRUCTURE CONCEPT

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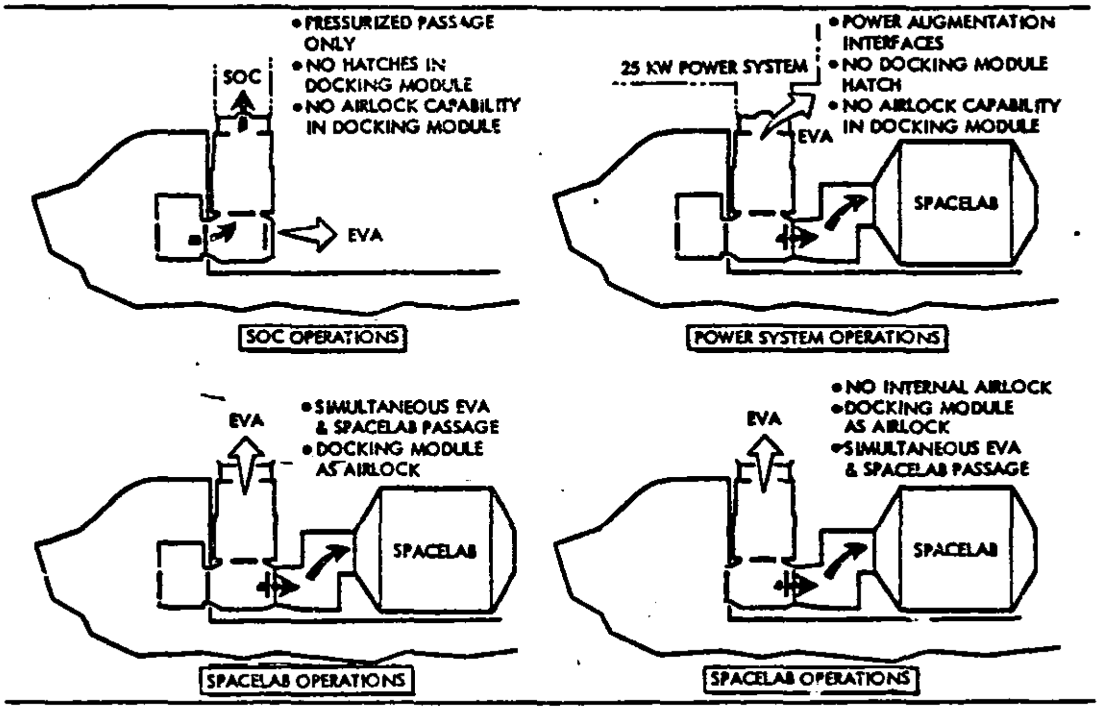

FIGURE 2.66 EVA OPTIONS

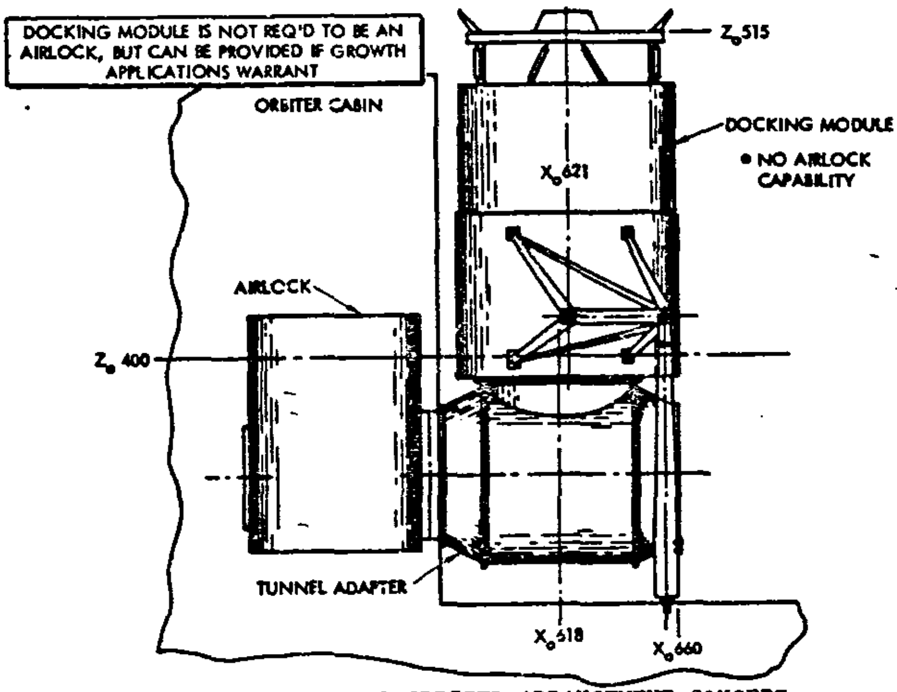

FIGURE 2.67 NOMINAL ORBITER ARRANGEMENT CONCEPT

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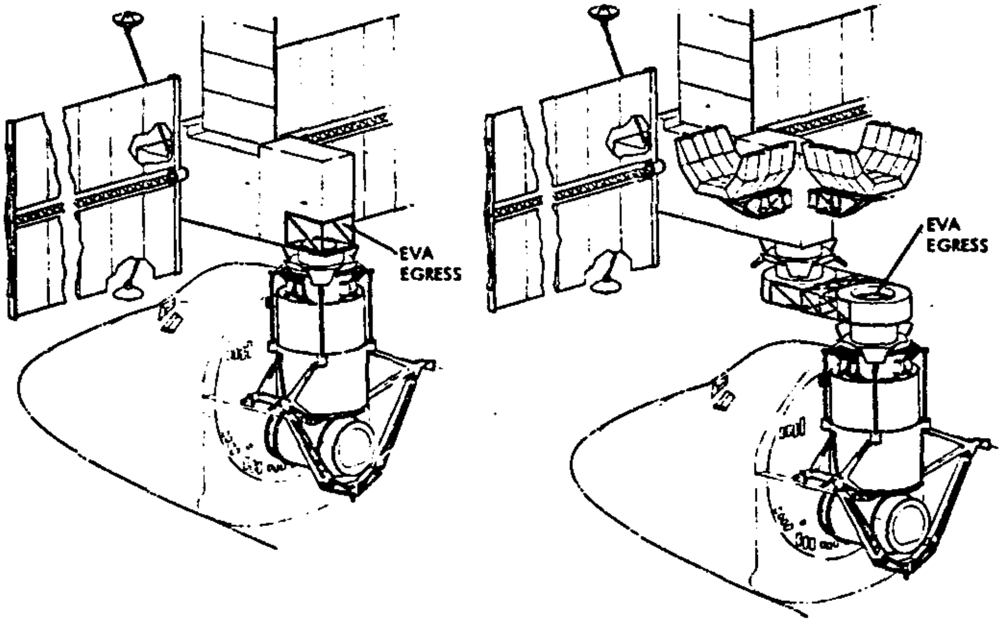

FIGURE 2.68 EVA DESIGN IMPLICATIONS FOR UNMANNED SYSTEMS

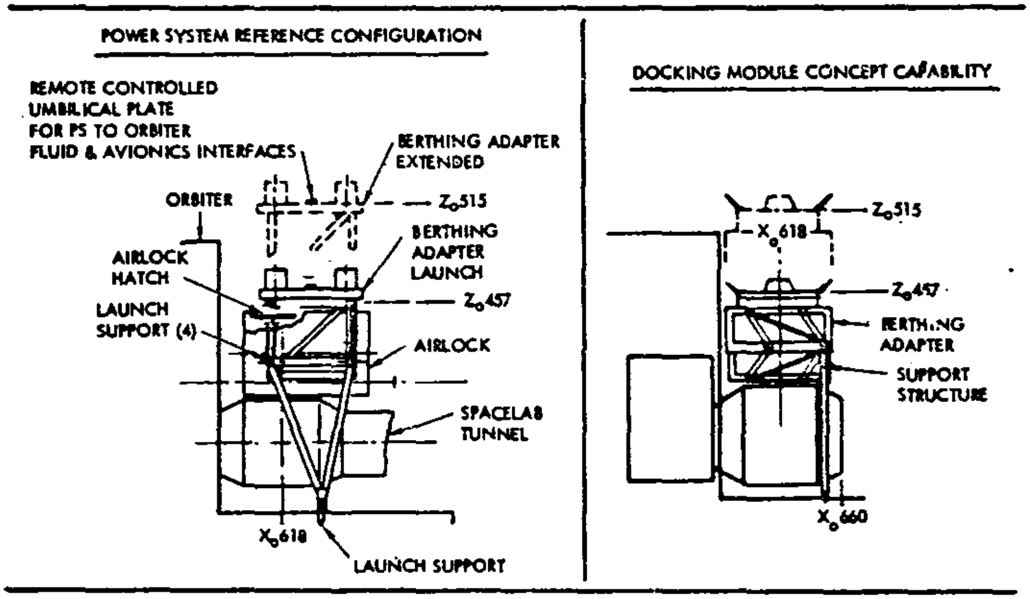

FIGURE 2.69 ALTERNATIVE DOCKING SYSTEM FOR UNMANNED PERATIONS

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TABLE 2.18 DOCKING MODULE REQUIREMENTS FOR STANDARDIZED CONCEPT

DOCKING MECHANISM

- THE DOCKING MECHANISM SYSTEM SHALL BE ANDROGYNOUS AND ALLOW DOCKING AT 90-DEGREE ALIGNMENT INCREMENTS.
- ALL DOCKING SYSTEM HARDWARE, EXCEPT THE ATTENUATORS, SHALL BE LOCATED WITHIN THE PRESSURE-SEALED INTERFACE AND BE ACCESSIBLE TO THE CREW FROM WITHIN THE PRESSURIZED ENVIRONMENT. THE IMPACT ATTENUATORS WILL BE LOCATED OUTSIDE OF THE PRESSURIZED INTERFACE AREA.
- A ONE-METER CLEAR OPENING WILL BE MAINTAINED THROUGH THE DOCKING MECHANISM AT ALL TIMES.
- THE GUIDE PETALS WILL BE ORIENTED ALONG THE ORBITER COORDINATE AXIS.
- DOCKING—DESIGN IMPACT CONDITIONS

- AXIAL CLOSING VELOCITY	0.015-0.15 m/sec (0.05-0.50 ft/sec)
- LATERAL VELOCITY	<0.06 m/sec (<0.2 ft/sec)
- ANGULAR VELOCITY	1.0 deg/sec (ABOUT ANY AXIS)
- LATERAL MISALIGNMENT	0.23 m (0.75 ft)
- ANGULAR MISALIGNMENT	<5.0 deg, ROLL; <6.0 deg, PITCH/YAW

- THE DOCKING MECHANISM WILL BE CAPABLE OF PERFORMING ACTIVE OR PASSIVE DOCKING/BERTHING

UTILITIES INTERFACES

- ALL UTILITIES INTERFACES SHALL BE LOCATED WITHIN THE PRESSURE-SEALED INTERFACE AND WILL BE REMOTELY ACTIVATED.
- ALL OF THE UTILITIES INTERFACES WILL BE ACCESSIBLE TO THE CREW FROM WITHIN THE PRESSURIZED ENVIRONMENT.
- ANY SERVICE INTERCONNECTIONS REQUIRED TO BE EXTERNAL TO THE PRESSURIZED AREA WILL BE UNIQUE FOR A PARTICULAR MISSION.
- THE UTILITIES SERVICES TO BE ACCOMMODATED ACROSS THE DOCKING INTERFACE ARE TBD.
- THE LOCATIONS OF THE UTILITIES INTERCONNECTIONS TO THE ORBITER SYSTEM ARE TBD.

TABLE 2.18 DOCKING MODULE REQUIREMENTS FOR STANDARDIZED CONCEPT (CONT.)

PRIMARY STRUCTURE

- EXTENSION TUNNEL & DOCKING MECHANISM WHEN RETRACTED WILL PROVIDE EVA CLEARANCE WITH THE PAYLOAD DOORS CLOSED (MIN. CLEARANCE, 36 IN. FROM DOCKING INTERFACE $-Z_O$ 457).
- EXTENSION TUNNEL WITH DOCKING MECHANISM WILL EXTEND 15 IN. BEYOND ORBITER MOLD-LINE (DOCKING INTERFACE AT Z_O 515).
- THE FIXED STRUCTURE OF THE EXTENSION DEVICE WILL BE SUPPORTED FROM THE ORBITER PAYLOAD BAY LONGERON BRIDGE STRUCTURE & CENTERLINE KEEL STRUCTURE.
- ACCOMMODATION OF ONE MMU/FSS ON THE LEFT SIDE OF THE ORBITER WILL BE PROVIDED.
- A FLEXIBLE, PRESSURE-SEALED JOINT WILL BE PROVIDED BETWEEN THE EXTENSION DEVICE & THE TUNNEL ADAPTER ELEMENT—NO DOCKING LOADS WILL BE TRANSFERRED TO THE TUNNEL ADAPTER.
- HATCH ACCOMMODATIONS WILL BE PROVIDED AT THE DOCKING INTERFACE END OF THE EXTENSION TUNNEL.
- AN EMERGENCY SEPARATION DEVICE WILL BE PROVIDED SUCH THAT SUBSEQUENT PAYLOAD BAY DOOR CLOSURE IS POSSIBLE. EMERGENCY SEPARATION SHALL NOT RESULT IN LOSS OF CABIN PRESSURE, UNACCEPTABLE VEHICLE DAMAGE, OR LOSS OF EVA CAPABILITY.
- AIRLOCK CAPABILITIES ARE NOT REQUIRED IN THE RETRACTED OR EXTENDED TUNNEL CONFIGURATION.
- A MINIMUM CLEAR PASSAGE OF ONE METER WILL BE PROVIDED THROUGH THE TUNNEL IN THE RETRACTED OR EXTENDED ARRANGEMENT.
- THE EXTENSION DEVICE WITH THE DOCKING MECHANISM WILL BE LOCATED BETWEEN X_O 582 AND X_O 660.

2.4 RMS BERTHING ANALYSIS

In addition to the preceding analyses related to direct docking of the Orbiter to the SOC the general feasibility of berthing the Orbiter to the SOC with the RMS was explored. Seven simulation runs were performed by SPAR Aerospace Limited of Canada, using their ASAD non-real-time simulation program. The purpose was to assess the general capability of the RMS for handling the very large system masses associated with the SOC and the fully loaded Orbiter and to identify any problem criticalities or important issues that might exist.

The SPAR simulation program is a high fidelity model which includes the effects of arm flexibility, joint freeplay dynamics and the current joint control software which was designed to handle payloads up to 65,000 lb. The simulation model did not include Orbiter or SOC body flexibility effects nor environmental disturbance torques (aero & grav-gradient).

Study emphasis was on analyzing the ability of the RMS to arrest the initial relative motion between the Orbiter and the SOC, but the subsequent use of the RMS to reposition the orbiter to a preberth position directly over the SOC docking port was also briefly investigated. A short description of the "problem model" used in the berthing analysis is presented below along with a summary of the simulation runs and related results. Appendix E contains the complete detailed analysis package with all the parameter plots as prepared by SPAR.

2.4.1 Physical Model for RMS Berthing Analyses

The reference configurations used in the simulation analysis and their mass properties are shown in Figures 2.70 and 2.71 for the SOC and the Orbiter, respectively. The SOC configuration is the JSC baseline consisting of the central cluster of basic SOC modules. Space construction and flight support facility elements are omitted except for the stage assembly module (SAM) which is part of the flight support facility. Also, no construction projects and/or OTVs and planetary stages are included. The addition of these elements, particularly a fully fueled MOTV could easily double or even triple the SOC mass properties listed here. The SOC cg and the location of the end effector grapple fixture are shown in SOC coordinates in Figure 2.70. As indicated by these data the grapple point is offset from the SOC cg by some 3 or 4 feet in the Y and Z directions and nearly 34 feet in the X direction.

The Orbiter configuration as depicted in Figure 2.71 is representative of an operational "bird" with the maximum 65,000 lb payload installed. In this heavily loaded condition, the Orbiter cg is offset from the RMS mounting location by nearly 38 feet.

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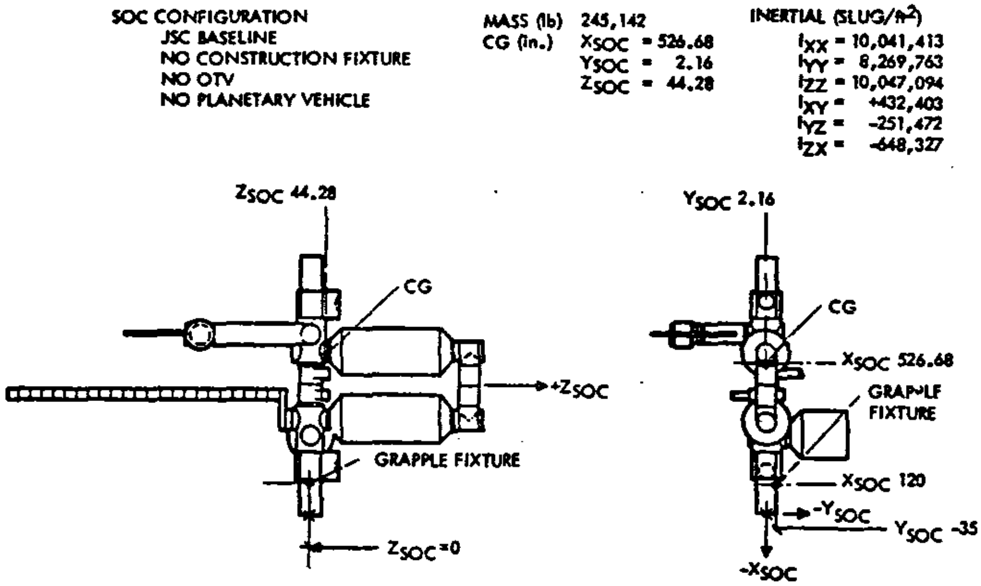

FIGURE 2.70. SOC MASS PROPERTIES FOR BERTHING ANALYSIS

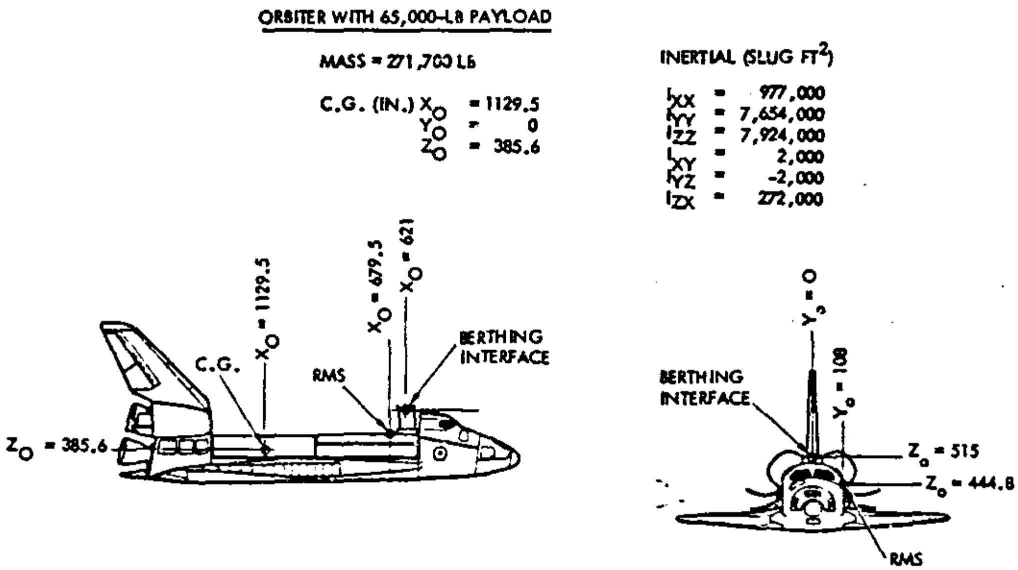

FIGURE 2.71. ORBITER MASS PROPERTIES FOR BERTHING ANALYSIS

The initial geometry for the berthing simulation runs is shown in Figure 2.72. The SOC berthing port was assumed to be centered directly over the Orbiter docking port and with SOC X-axis perfecty aligned with the Orbiter Z-axis (no angular misalignment). The separation distance was maximized within "comfortable" RMS reach limits which was initially estimated to be the 24.8 feet shown. It was later determined that this distance must be reduced by 2 or 3 feet to assure that shoulder and elbow pitch joint limits are not exceeded by the berthed condition. This adjustment was not incorporated into the sim runs, but was felt to have negligible study impacts. This was born out by the simulation results which showed maintaining a safe separation distance was not a critical factor. The arm geometry at the point of end effector engagement is shown in the figure along with the initial values of all joint angles.

2.4.2 Simulation Results Summary

Seven simulation runs were generated, 5 to study the RMS capability to arrest residual motions and 2 to explore its ability to maneuver the Orbiter to the SOC port once the residual motions were nulled. Of the 5 motion arrest cases 2 were aimed at determining if the direction of relative motion was of any significance, one was to determine the effects of software mods on arm stability, one was to probe the effects of berthing to a stabilized SOC (active altitude control) and one to determine if angular residual motions posed greater problems than linear motions. The two arm maneuvering runs to position the Orbiter over the SOC port were aimed at the capability differences between manual and automated arm positioning modes. The overall results are summarized in Table 2.19. Individual case conditions and their respective results are presented. The general finding is: RMS berthing is feasible, but requires minor mods to the control software. Large residual motions (compared to expected values) can be nulled within stopping distances and angles of 18 inches and 5 degrees.

The data from run 1, Figure 2.73, illustrates the need for software mods to acheive stable control. The current software is shown to produce undamped oscillations using the manual augmented mode (MAM) to arrest residual motions. Run 3 in Figure 2.74 shows stability is achieved for the same case conditions, but with the MAM modified to eliminate the automatic switchover to the position hold (PH) submode which occurs when all joints are within the 0.05 deg/sec rate threshold. The higher control gains inherent within this submode produced the instability exhibited in run 1.

Run 4 in Figure 2.75 shows the modified (MAM) mode provides stable control for the case with a stabilized SOC. All other sim runs were made with the Orbiter and the SOC in free drift modes (no attitude control). Somewhat higher frequencies are seen in the arm response characteristics for this case, but stable control is exhibited. Active SOC attitude control was emulated in this case by applying a 10^7 factor to the SOC moments of inertia (mass was held the same as for the other runs). This does not precisely simulate all of the control system dynamics that would be present in the actual SOC attitude control system, but is does give a feel for the principal effects of a relatively tight SOC attitude control concept during the initial berthing operations (motion arrest).

TABLE 2.19 RMS BERTHING RESULTS SUMMARY

RUN	CASE CONDITIONS	SUMMARY RESULTS
1	• ARREST INITIAL MOTION (.1 ft/sec, .025°/sec) MOTION IN THE ARM PLANE • MAM/CONTROLLERS IN NEUTRAL (I.E., ZERO RATE COMMANDS)	• PHM AUTOMATICALLY ENGAGED FEW SECONDS AFTER "RIGIDIZATION" • MARGINAL STABILITY –NO APPRECIABLE DAMPING (800 sec) –SOC CENTER OF MASS PEAK-TO-PEAK EXCURSIONS 1.5 ft
2	• SAME AS ABOVE WITH INITIAL MOTION PERPENDICULAR TO THE ARM PLANE	• UNDAMPED OSCILLATION
3	• ARREST INITIAL MOTION (.1 ft/sec, .025°/sec) MOTION IN THE ARM PLANE • MODIFIED MAM/CONTROLLERS IN NEUTRAL	• STABLE CONTROL EXHIBITED AFTER [illegible]90 SECONDS –SOC CENTER OF MASS PEAK-TO-PEAK EXCURSION WITHIN 1 INCH –SOC ATTITUDE EXCURSION WITHIN 0.2 DEG • RELATIVELY HIGH LOADS FOR SHORT PERIOD IMMEDIATELY AFTER RIGIDIZATION; LEVELS ACCEPTABLE
4	• SAME AS ABOVE WITH SOC INERTIA 10^7 HIGHER THAN BASELINE; SIMULATE "STOPPING PHASE" WITH SOC ACS ACTIVE	• HIGHER FREQUENCIES ARE EXHIBITED AND SLIGHTLY HIGHER LOADS, BUT STILL WITHIN ACCEPTABLE LEVELS
5	• MANEUVER SOC WITH MODIFIED MAM • INITIAL CONDITIONS FROM END OF RUN 3 • COMMAND TOWARDS "PREBERTH" POSITION/ORIENTATION	• SUITABLE STRATEGY FOR MANEUVERING THE SOC
6	• USING SLIGHTLY MODIFIED OCAS MODE –MANEUVER THE SOC TO "PREBERTH" POSITION/ ORIENTATION –STABILIZE THE SOC AT "PREBERTH" • INITIAL CONDITION FROM END OF RUN 3	• OCAS QUITE SUITABLE FOR MANEUVERING THE SOC • OCAS NOT SUITABLE FOR STABILIZING THE SOC; MARGINAL STABILITY IS EXHIBITED NEAR THE "PREBERTH" POSITION
7	• ARREST HIGH ANGULAR MOTION (.052 ft/sec, .1732°/sec) • MODIFIED MAM/CONTROLLERS IN NEUTRAL	• MODIFIED MAM CONFIRMED AS THE STRATEGY FOR STOPPING AND/OR STABILIZING THE SOC

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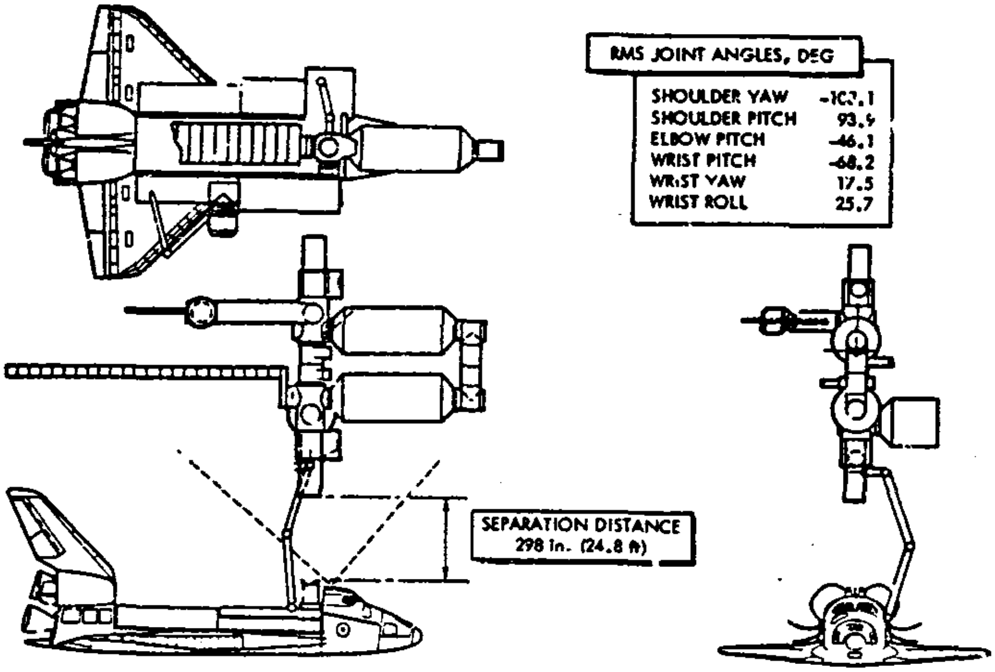

FIGURE 2.72. ORBITER/SOC BERTHING GEOMETRY

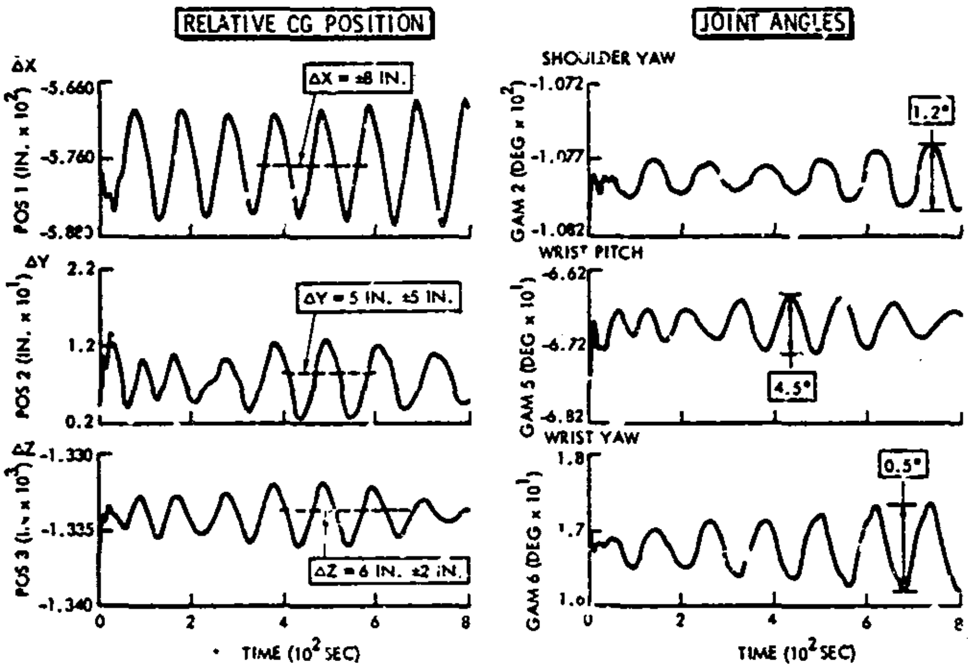

FIGURE 2.73. BERTHING SIMULATION RESULTS - RUN 1

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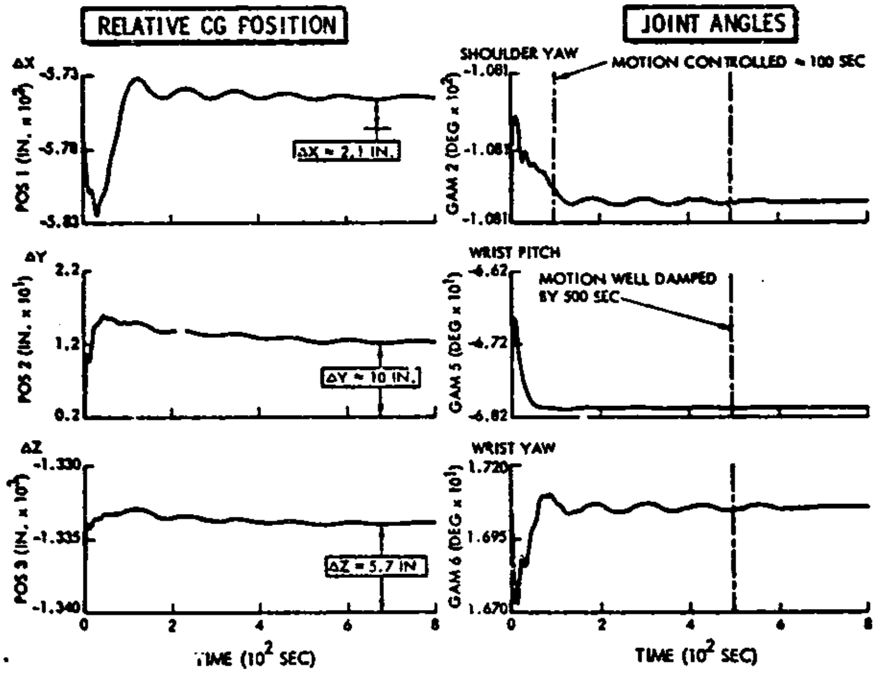

FIGURE 2.74. BERTHING SIMULATION RESULTS - RUN 3

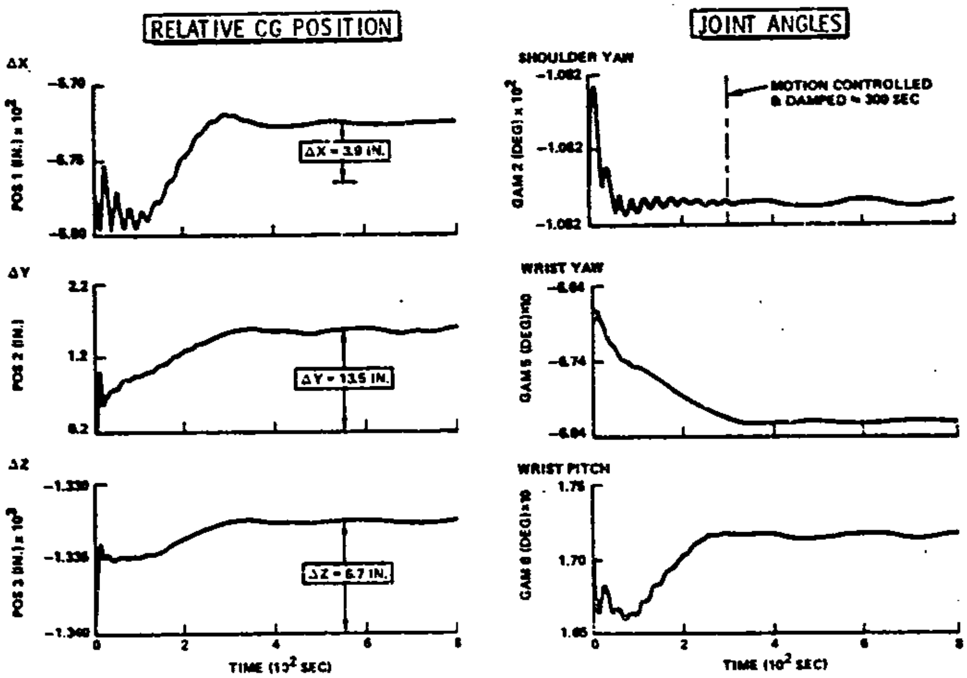

FIGURE 2.75. BERTHING SIMULATION RESULTS - RUN 4

As illustrated in run 5, Figure 2.76, the modified MAM could also be used to reposition the Orbiter to the SOC docking port. Stable control is produced. Significant drift was noted in the joint control electronics, but with suitable video aids the RMS operator should be able to manually control the Orbiter to the desired berthing position. Settling time interactions with these joint drift rates may cause some difficulties. Real time, man-in-the-loop simulations are required to determine the significance of these interactions.

Stability problems were encountered in run 6. As illustrated in Figure 2.77, further mods to the control software would be required to produce stable control with the automated arm maneuvering modes. With the automated modes, a "washout zone" is used to smooth the stopping action at the designated position. Within the washout zone (several feet from the target) distance and angle to go data are calculated and fed into the arm control network to insure a smooth transition to the desired final position and orientation. This feedback function in the automated arm maneuvering modes has the same effect as the position hold submode in the motion arrest cases above. The effective control gains are increased and instability results.

Run 7, Figure 2.78, shows that the effects of high angular residual motions instead of high linear motions introduce no special problems. Stopping distances are small and stable control is provided by the modified MAM.

It is concluded that modifying the RMS control software to eliminate the instabilities, judged by SPAR to be minor changes, will result in satisfactory berthing performance. The simulations indicated wide separation margins between the Orbiter and the SOC can be maintained and peak arm loads, while high will be within their design limits. Thus, RMS berthing the Orbiter to the SOC is deemed feasible, but requires minor modifications to the arm control software.

Further analyses are required to determine (1) Orbiter/SOC flexible body effects, (2) effects of increased mass properties for SOC growth configurations and uprated/augmented thrust Orbiter concepts and (3) the most effective software changes applicable to the range of conditions expected for the overall SOC scenario.

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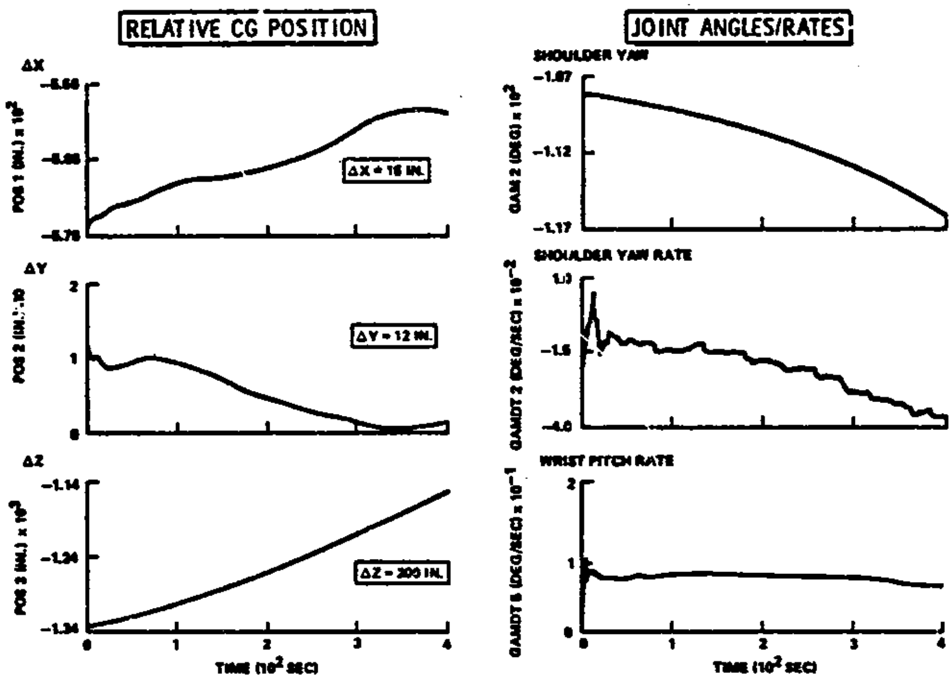

FIGURE 2.76. BERTHING SIMULATION RESULTS - RUN 5

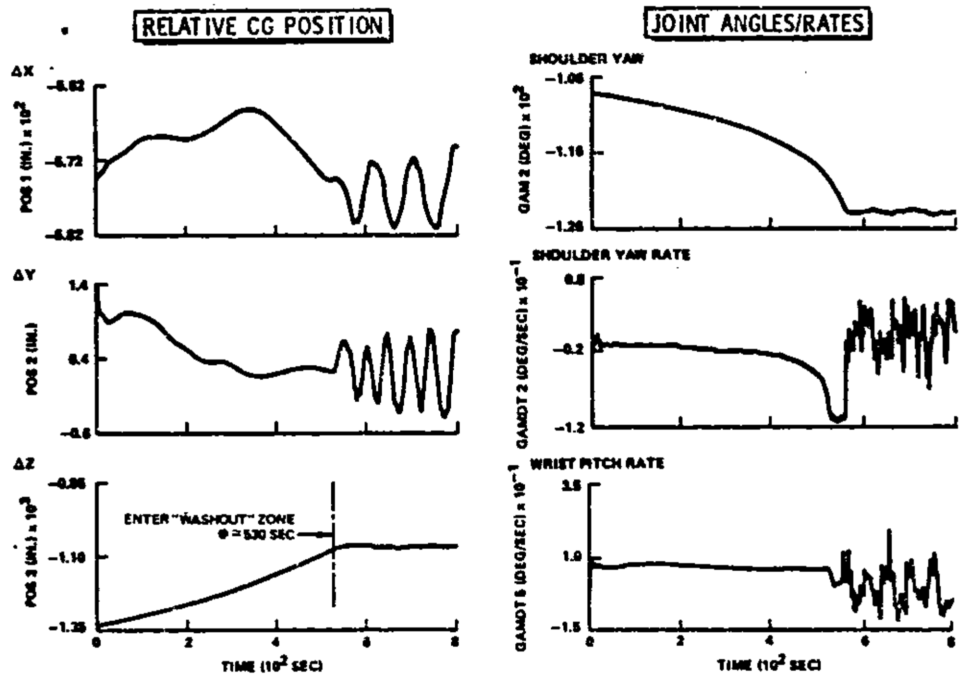

FIGURE 2.77. BERTHING SIMULATION RESULTS - RUN 6

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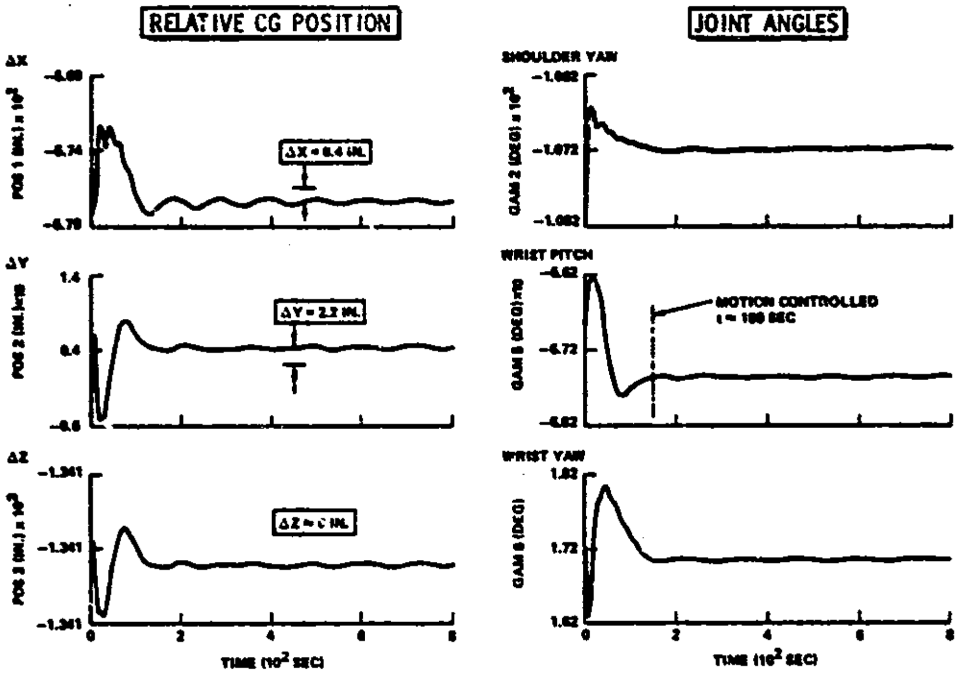

FIGURE 2.78. BERTHING SIMULATION RESULTS - RUN 7

References

1. Hoffman, R. J, et.al; Space Shuttle RCS Plume Contamination Study, Report No. MDC G5250, Contract P. O. M3W3XM2-483001, McDonnell Douglas Astronautics Company - West, Huntington Beach, California, July 1974

2. IL SAS/MR&I-77-167B from B. C. Johnson to H. E. Emigh; Subject: "Orbiter Plume Data for Shuttle Vehicle/Cargo Standard Interface Specification", dated 1 September 1977

3. IL SAS/MR&I-77-187 from B. C. Johnson to H. E. Emigh; Subject: "Core-ICD Plume Data for the Revised VRCS Engine Thruster Performance", dated 21 September 1977

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5. French, E. P.; Rocket Plume Effects in Space, SD74-SA-0137, Rockwell International, Space Division, Downey, California, October 1974

6. Baerwald, R. K.; Rocket Exhaust Plume Impingement on the Voyager Spacecraft. AIAA Paper 78-1090, presented at the AIAA/SAE 14th Propulsion Conference, Las Vegas, Nevada, 25-27 July 1978; American Institute of Aeronautics and Astronautics, N.Y., N.Y.

7. SD 72-SH-0105; Requirements/Definition Document -- Flight Control; Vol. 1, Book 2, Part IB, dated February 1980.

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3.0 SOC ASSEMBLY

The SOC assembly procedure is premised on the reference configuration depicted in Figure 3.1 and the use of the Shuttle Orbiter as an assembly base. In that capacity, the role of the orbiter, the requirements imposed on it, and the equipment needed to perform the assembly operations must be determined. In this section of the report, these issues are examined in conjunction with several assembly scenarios. In addition, visibility, lighting and CCTV provisions and the requirements to stabilize the untended SOC during partial assembly configurations are presented.

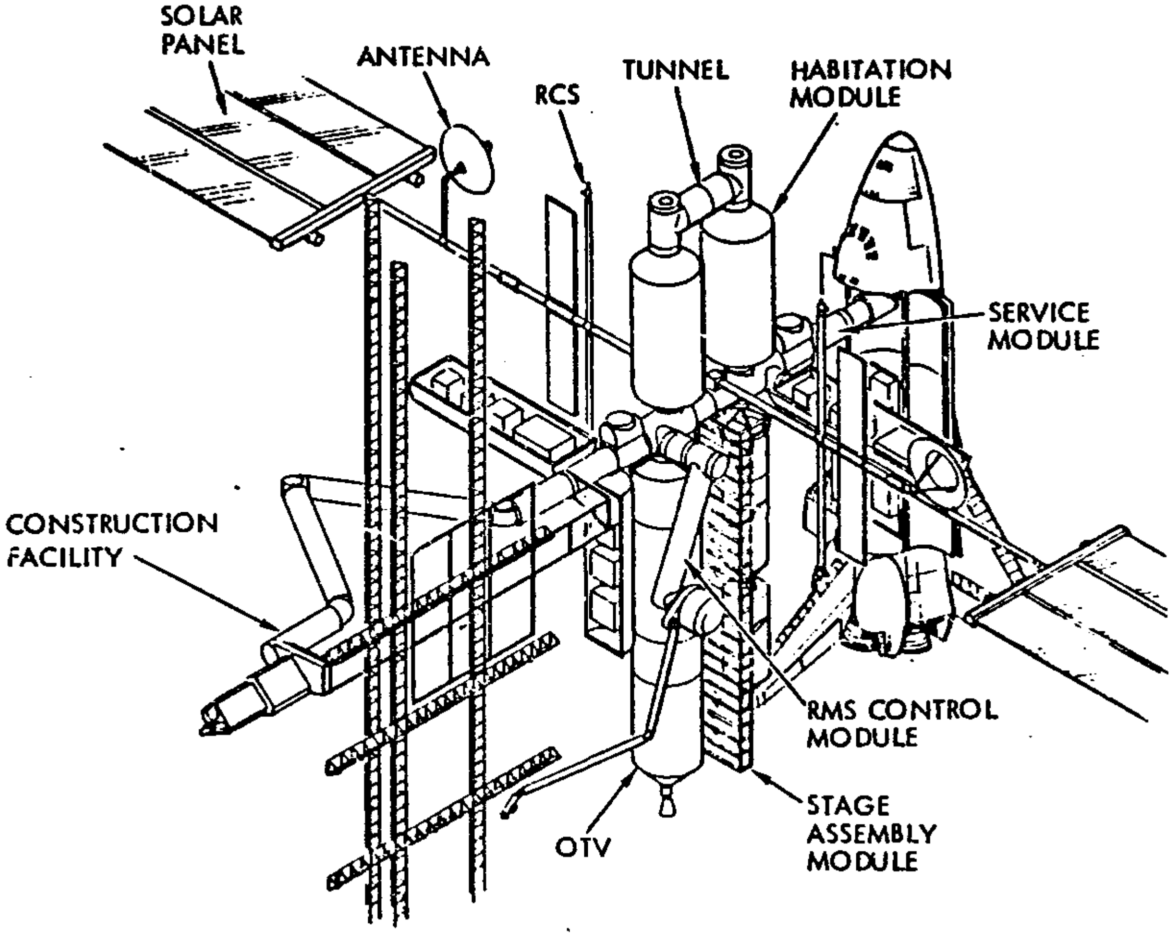

FIGURE 3.1 SOC REFERENCE CONFIGURATION

3.1 SUMMARY

The SOC Can and Should Be Designed to Accommodate Variations in the Build Plan

Future uncertainties lead to the need for versatility in the SOC design. The exact nature, timing and rate of growth of user program requirements are difficult to predict. There are also uncertainties in funding availability. Program funding is subject to other national priorities and can vary with general economic trends. Further, international space developments may introduce new urgencies and/or changes to near term program emphasis. Thus, the SOC design concept should allow for variations in the buildup scenario to meet these evolving future needs and influences.

Preliminary analyses have shown that modest changes to the baseline SOC configuration can satisfy a wide range of build plans. For example, the baseline RCS system is not adequate for full functional capability with just one service module. A small supplemental RCS system will satisfy the control needs for the direct buildup scenario, but would be seriously limited for most evolutionary build plans. However, changing the design to allow installation of the complete baseline "dual boom" RCS system on the first service module (instead of half a system on each service module) would satisfy the RCS control functions and redundancy needs for virtually all buildup options.

Also, momentum control concepts sized to the full up SOC configuration can meet the needs of all early mission options. Even assuming one-half of the momentum control is located in each service module there is enough capacity to perform many early missions with configurations centered on the use of a single service module.

<u>SOC Assembly Variations</u>

In accordance with guidelines based on a NASA-generated buildup sequence, a baseline scenario was generated. In the baseline scenario, the SOC modules are produced and delivered in the most logical sequence to achieve "full-up" operational capability in the earliest feasible time period. Two variations to the baseline were examined in which the delivery of the RMS Control Module was manipulated to assess its sensitivity to the overall construction of the SOC. As a consequence of this analysis, a handling and positioning aid (HPA) was identified as a necessary piece of equipment, by one of the variations, in order to achieve the overall SOC assembly. The HPA was also a significant factor in generating two evolutionary SOC assembly scenarios that provide a high degree of mission flexibility. The desirability of an evolutionary approach would be further enhanced by a build sequence that permits the earliest operational capability and, at the same time, minimizes front end costs.

The analysis resulted in a high confidence technique for the assembly of the SOC modules. It was further indicated that the SOC can be designed to accommodate nominal and evolutionary buildup modes, and variations to those modes with the addition of relatively minor provisions.

SOC Assembly Visibility

An analysis of lighting and TV requirements for the initial assembly operations of the SOC was conducted to determine what features should be incorporated into the SOC and to define potential orbiter impacts. The SOC impacts resulting from the analysis included a concept for installation of a TV camera and a light (or lights) on one side of each berthing port, with a target on the mating side (up coming module). This concept is shown in Figure 3.2. In addition, the recommendation includes four exterior flood lights on the RCM cabin, a movable (tilt and pan/TV camera and light set on the RCM manipulator wrist and elbow, and approximately 45 small, colored marker lights distributed around the extremities of the SOC. The latter includes four marker lights on each of three orbiter docking ports.

Recommended minimum changes to the orbiter include a movable lamp on the aft bulkhead of the cargo bay, two movable TV cameras and lights on the aft sides of the docking module exterior and a TV camera and light(s) in the docking port. The proposed locations for all lights, cameras targets are summarized in Figure 3.3. Note that if a handling and positioning aid is used as proposed in some alternate schemes, a TV camera and light are also recommended for the attach port.

The peak power requirements estimated for the orbiter cargo operations could be as high as 4.5 to 6.1 kW, while the SOC peak power for lighting and TV near the end of the buildup sequence could require up to 6.6 kW. It was presumed that SOC power would be available to support its own lighting, TV and RCM operations from (at least) a partially deployed solar array after the first flight in the assembly sequence.

Other design features which can enhance the versatility and enlarge the range of early SOC applications are the use of independent life support systems in the two habitability modules and the incorporation of "docking" capability in the tunnel assembly for joining habitable volumes.

Thus, it is concluded that the SOC design concept can and should have the versatility to accommodate a wide range of evolutionary build plans.

3.2 SOC ASSEMBLY SEQUENCE VARIATIONS

The use of the Shuttle Orbiter as a base for space construction has been thoroughly studied by many industry and NASA investigations. The thrusts of most of these studies dealt with construction of space platforms from basic building block structural elements. Very little emphasis was placed on assembling large modules together as in the SOC. Nevertheless, there were indications that the orbiter and its support equipment can accomplish the task. To investigate the feasibility of such an approach, a logical SOC assembly sequence must be determined. In this portion of the report, descriptions of several assembly scenarios are presented with emphasis on the geometric configuration of the SOC during its assembly.

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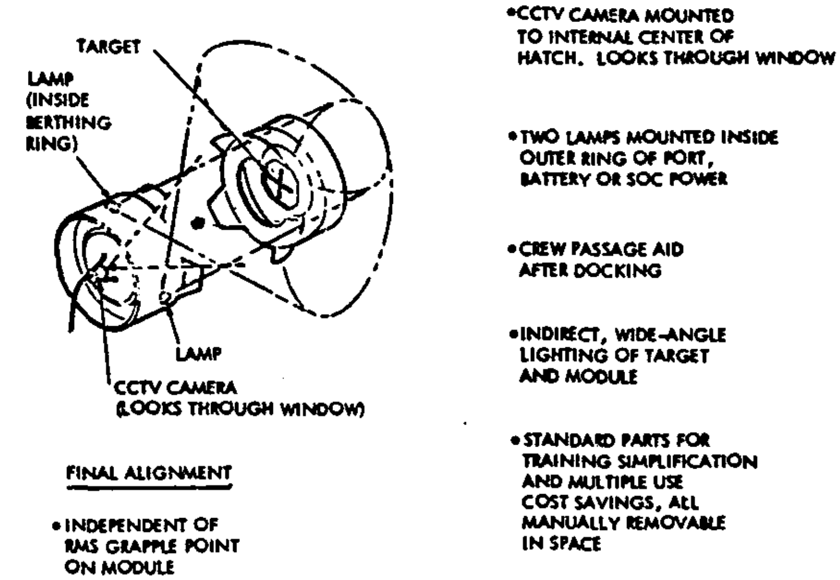

FIGURE 3.2 CONCEPT FOR BERTHING PORT LIGHTING/TV AND TARGET KIT

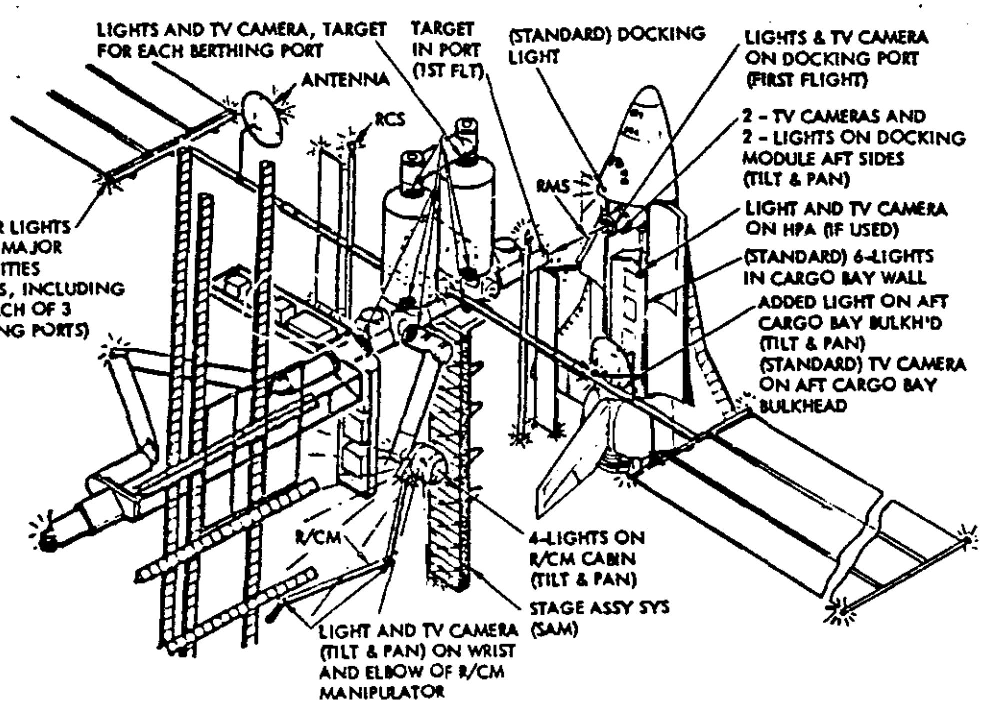

FIGURE 3.3 LIGHTING AND TV CAMERAS FOR SOC ASSEMBLY

In generating the scenarios, a number of operational issues were considered for assembling the SOC. Table 3.1 lists the issues along with the various considerations that were specifically addressed in relation to the issues. Two of these issues, alignment aids and stabilization of the untended SOC assembly, were found to be extremely significant and an in depth treatment of both issues are presented in paragraphs 3.3 and 3.4. As a further aid in formulating the scenarios, 1/48 scale models of the orbiter and the various SOC modules were utilized to simulate the buildup sequences. The fidelity of the models was sufficient to establish the feasibility of the selected SOC-orbiter orientations and to determine gross reach capabilities of the manipulators utilized in the assembly. A photographic array of the 1/48 scale models is shown in Figure 3.4 in which a variation to the baseline sequence is illustrated.

A significant result of the SOC assembly task was the identification of two major pieces of assembly equipment which facilitate the SOC assembly. These pieces of equipment, the Payload Installation and Development Aid (PIDA), and the Handling and Positioning Aid (HPA) are in addition to those considered standard equipment aboard the orbiter and the SOC. The PIDA, Reference 1, is intended to move large payloads through a prescribed and controlled path between the confined quarters of the payload bay and a position outside the critical maneuvering area of the orbiter. As utilized in the SOC assembly, two synchronized drive PIDA arms deploy a SOC module out of the payload bay in a two-stage movement as illustrated in Figure 3.5. Each PIDA arm consists of a deploy/stow mechanism, a payload interface mechanism, an electromechanical rotary actuator with its respective electronic controls, and a base, with a jettison interface, that connects the assembly to the orbiter longeron bridge fitting as shown in Figure 3.6.

The HPA is mainly an attachment device to which various payloads can be berthed and which provides a multi-positioning/orientation capability of the various payloads for orbiter-based construction and near-orbiter satellite servicing activities. In that capacity, it complements the role of the RMS by maximizing its reach and enhancing its capability to access points on payloads of complex geometric configurations that otherwise would be inaccessable to the RMS as illustrated in Figure 3.7. The HPA also provides improved visibility of the particular work site to the RMS operator or brings the work site within the direct line of sight of a CCTV camera. Conceptually, the HPA is envisioned as a relatively stiff arm with five degrees of freedom. It will consist of an orbiter interface mechanism, a tubular shaped drive arm and a payload interface mechanism with its own electronic control as seen in Figure 3.8.

3.2.1 SOC Assembly - Baseline Scenario

The baseline assembly sequence utilized the NASA/JSC referenced sequence. A detailed analysis of this sequence was conducted which identified all required berthing/docking operations, examined translation paths, reach distances and visibility conditions. To fully assemble the SOC by the baseline scenario, seven orbiter flights are required to transport

TABLE 3.1 SOC ASSEMBLY OPERATIONAL ISSUES

ISSUE	CONSIDERATION
ORIENTATION OF STOWED SOC MODULE	• COMPATIBLE FOR DEPLOYMENT WITH PIDA • COMPATIBLE FOR GRASPING WITH RMS • COMPATIBLE WITH FINAL ASSEMBLED POSITION
BERTHING PORT SELECTION	• COMPATIBLE WITH RMS REACH • VISIBILITY—LIGHTING
TRANSLATION PATH	• COMPATIBLE WITH RMS ARTICULATION CAPABILITY • CLEAR ORBITER APPENDAGES • CLEAR SOC APPENDAGES
ALIGNMENT AIDS	• MINIMAL IMPACT ON SOC MODULES • SIMPLE AND RELIABLE CONCEPT • VISIBILITY—DIRECT OR CCTV—LIGHTING
CHECKOUT	• PREDEPLOYMENT CHECKOUT OPERATIONS • POST-DEPLOYMENT CHECKOUT OPERATIONS
STABILITY OF UNTENDED SOC	• SOC PARTIAL ASSEMBLIES • AUXILIARY SYSTEMS

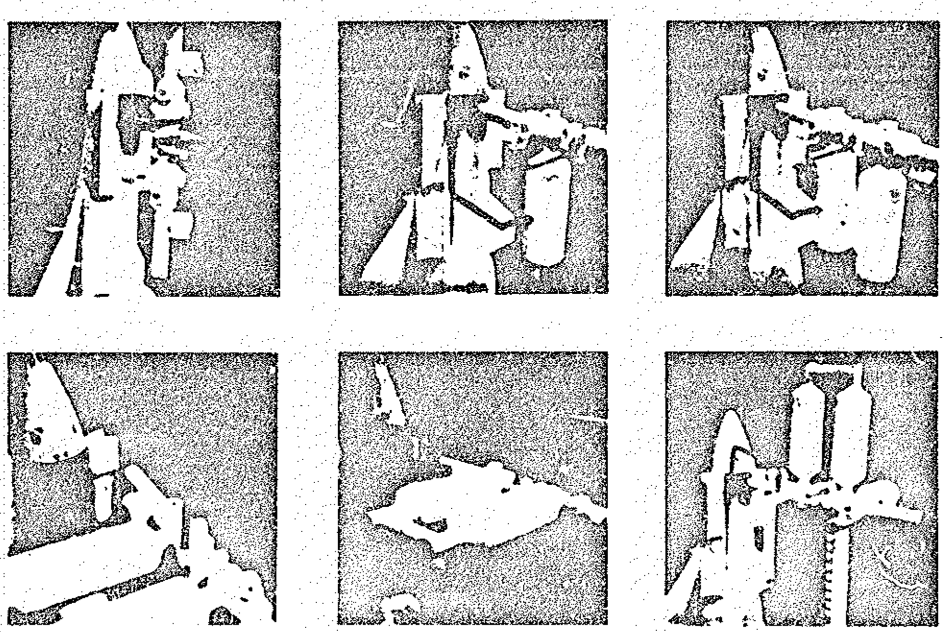

FIGURE 3.4 BUILDUP EVALUATION TECHNIQUE

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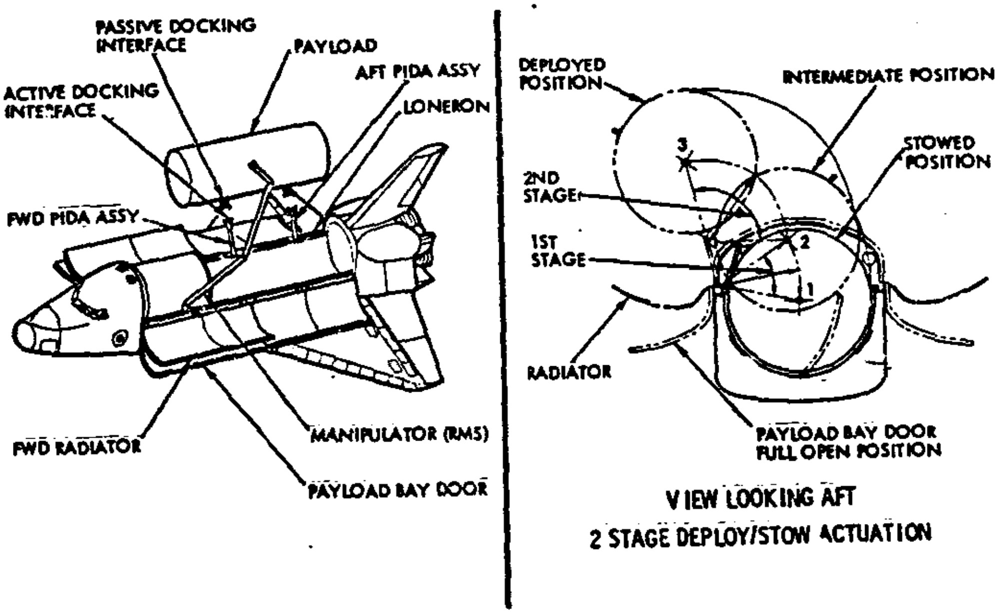

FIGURE 3.5 PIDA OPERATION

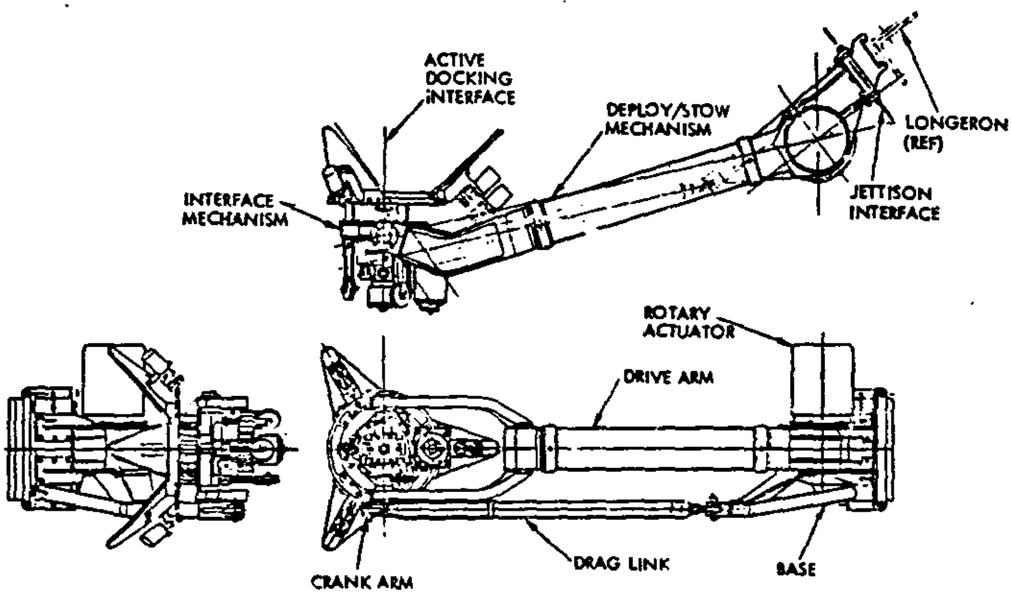

FIGURE 3.6 PIDA ASSEMBLY

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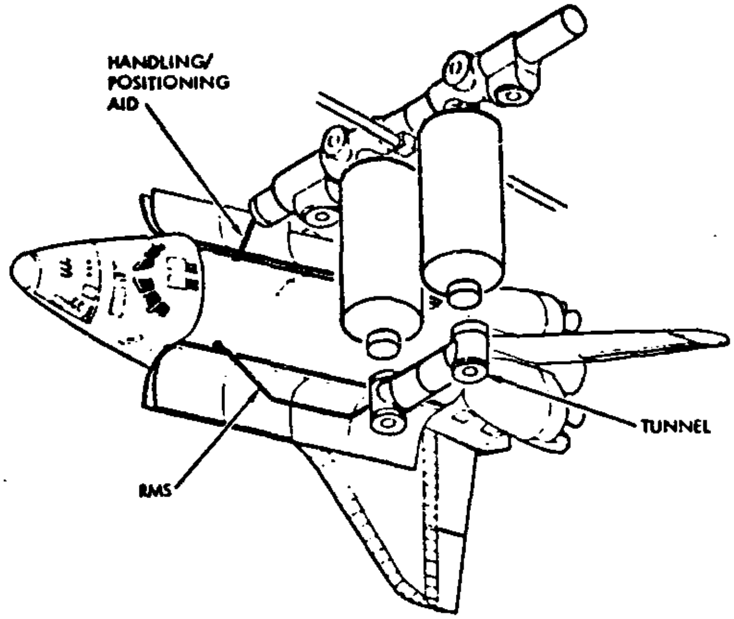

FIGURE 3.7 EXAMPLE OF SOC ASSEMBLY OPERATION USING HPA

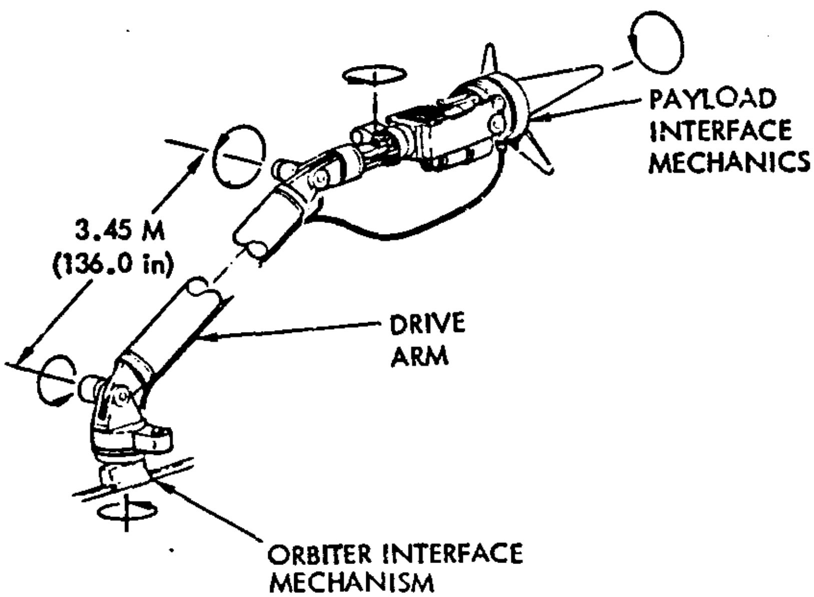

FIGURE 3.8 HANDLING AND POSITIONING AID CONCEPT

all of the SOC modules. The payloads for each of the seven flights and sequence of their assembly are listed in Table 3.2 and illustrated in Figure 3.9. Each of the flights is described separately along with the major operations required to attach its payload to the SOC and some of the developmental issues that will require further investigations.

Baseline Scenario - Flight No. 1

Payload: Service Module No. 1 (SM-1)

Equipment: Payload Installation and Deployment Aid (PIDA)
Remote Manipulator System (RMS)
Docking Module (DM)

Once the assembly orbit is achieved, SM-1 is deployed from the orbiter payload bay using the PIDA. SM-1 is then grappled by the RMS, which transports it to the vicinity of the DM, orients it and effects a berthing operation between the end port of SM-1 and the DM as shown in Figure 3.9. A checkout operation of SM-1 follows including solar array deployment and retraction.

It is desirable to arrange the stowage of SM-1 so that the PIDA heads would interface with the main body of SM-1. Consequently, the SM-1 airlock needs to be oriented down in the payload bay and the body up so it can be closer to the PIDA heads. With that position, the orientation of SM-1 on the DM with the airlock pointing aft appears to be the easiest for the RMS to perform and was adopted as the baseline orientation.

A complete checkout of SM-1 prior to its deployment from the payload bay cannot be performed, e.g., deployment of the solar array. Consequently, it was established that only hazard red lines, if any, need to be monitored for each module while awaiting deployment from the payload bay. On that basis, a complete checkout of SM-1 can only be performed after it is berthed with the DM.

At the conclusion of the checkout operation, the analysis identified the need to establish an untended SM-1 configuration. Specifically, the control stability of SM-1 with or without a deployed solar array boom must be determined to establish flight approaches for orbiter revisits. This condition exists after each flight until the SOC is fully assembled. Consequently, requirements for the untended SOC were derived and are presented in Section 3.4.

Baseline Scenario - Flight No. 2

Payload: Service Module No. 2 (SM-2)

Equipment: PIDA, RMS & DM

The discussion of Flight No. 1 regarding stowage within the payload bay and deployment also pertain to Flight No. 2 since the payloads are similarly configured. The distinguishing feature of this flight is the selection of a

TABLE 3.2 SOC ASSEMBLY - BASELINE SCENARIO BUILDUP SEQUENCE

MODULE		CONFIGURATION						
		1	2	3	4	5	6	7
SERVICE MODULE 1	(SM-1)	X	Y	X	X	X	X	X
SERVICE MODULE 2	(SM-2)		X	X	X	X	X	X
HABITATION MODULE 1	(HM-1)			X	X	X	X	X
HABITATION MODULE 2	(HM-2)				X	X	X	X
STAGE ASSEMBLY	(SAM)					X	X	X
RMS/CONTROL	(R/CM)					X	X	X
TUNNEL	(TM)						X	X
LOGISTICS MODULE	(LM)						X	X
CONSTRUCTION FACILITY	(CF)							X

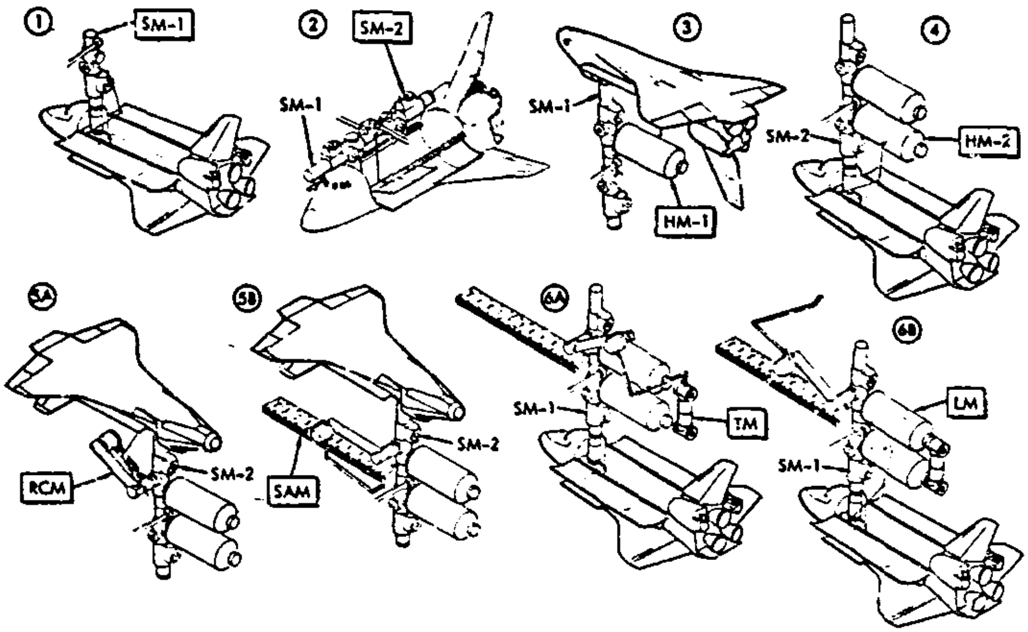

FIGURE 3.9 SOC ASSEMBLY - BASELINE SCENARIO

side port on SM-1 to interface with the orbiter DM. Interfacing the orbiter to the side port, as shown in Figure 3.9, was found necessary to bring the attachment operation of SM-2 to SM-1 within the useful reach of the RMS. As a consequence of the resulting arrangement, a berthing (rather than a docking) operation of the orbiter to SM-1 may be required when considering the close proximity between SM-1 and the crew cabin.

The visibility requirements of the RMS operator during all aspects of assembly operations were recognized early in the analysis. It is apparent that the RMS operator will be unable to have adequate visual contact with most docking/berthing interfaces. On-axis optical devices can provide adequate visibility and a discussion of their use is presented in Paragraph 3.3 along with lighting and CCTV provisions that are required during assembly operations.

Baseline Scenario - Flight No. 3

Payload: Habitation Module No. 1 (HM-1)

Equipment: PIDA, RMS & DM

The orbiter can be docked or berthed to the end port of SM-1 at the initiation of the assembly operation of this flight. Deploying HM-1 from the payload bay will require the use of a PIDA because of its large diameter. The RMS reach to accomplish the task as depicted in Figure 3.9 is adequate. To minimize translation distances and unnecessary articulation maneuvers on the part of the RMS, the HM-1 must be stowed so that its SOC interfacing port is facing forward while in the payload bay.

Baseline Scenario - Flight No. 4

Payload: Habitation Module No. 2 (HM-2)

Equipment: PIDA, RMS & DM

The operations that are required to transport, deploy and attach HM-2 to the SOC are similar to those of HM-1. The notable exception is the interface between the orbiter and the SOC assembly. In this case, the orbiter must dock or berth to the end port of SM-2 rather than SM-1 to bring the attachment operation of HM-2 to the SOC within the usable reach of the RMS. Since the RMS is the only tool available at this point in the assembly sequence to accomplish the task, it is necessary for the orbiter to interface with SM-2. It must be noted that this is the third separate orbiter interface port that was dictated by the baseline assembly sequence which leads to the issue of standardized orbiter/SOC terminal closure and docking maneuvers. Variations in approach paths and orbiter attitudes can effect the precision of the closing trajectory and the required amount of RCS thruster activity. It is imperative to minimize the number of interface ports so standardized docking maneuvers can be adopted to simplify the docking operations.

Baseline Scenario - Flight No. 5

Payload: Stage Assembly Module (SAM)
RMS/Control Module (RCM)

Equipment: RMS & DM

To minimize the need for undocking and redocking on this flight the RCM arm is proposed as an assembly tool. The procedure calls for the orbiter RMS to attach the RCM to the SOC initially. Then, by the aid of the RCM arm, the SAM is deployed from the payload bay and attached to the SOC. This procedure imposes a significant requirement, i.e., a crewman inside the SOC to operate the RCM arm. (This requirement applies to all assembly concepts wherever the RCM arm is used as an assembly tool). Crew transfers into the SOC were not required prior to this flight. If subsequent unforseen factors prohibit the use of the RCM arm at this time, then the orbiter RMS can be employed to attach the SAM by undocking the SOC/orbiter and redocking at the end port of SM-1.

As envisioned in the reference SOC configuration of Figure 3.1, the SAM was assumed to be a deployable module with a stowed configuration that will allow its stowage within the same orbiter as that carrying the RCM. If it is not a deployable module, as proposed in Section 5.0, one additional orbiter flight will be required to assemble it to the SOC, but it will not alter the operational assembly procedure significantly. When the SAM is assumed to be deployable, it is desirable to postpone its final deployment until the subsequent flight when the SOC can be declared operational. In this way, the operational flexibility of the RCM will be maximized by the absence of a deployed SAM appendage.

Detail design definition on both the SAM and the RCM were not available to investigate their packaging arrangement within the payload bay. To minimize the RCM stowage envelope, however, its cabin and interface tunnel need to be oriented in the same direction and, most likely, will be berthed to the SOC in that orientation.

Baseline Scenario - Flight No. 6

Payload: Tunnel Module (TM)
Logistics Module (LM)

Equipment: PIDA, RMS, DM & RCM

This is the only flight of the baseline scenario where the reach capabilities of RMS were found inadequate to attach the TM to the SOC while the SOC is attached to the DM. This condition exists regardless of scenario. For the baseline scenario, the RCM arm is utilized for the attachment of the TM and the RMS is utilized for the attachment of the LM. The RCM arm reach capability is adequate to deploy the TM from the payload bay and to access both ends of the TM for attachment to the SOC. The RCM arm is inadequate for the attachment of the LM to the SOC. This condition was verified by the 1/48 scale model. Consequently, the LM berthing task was assigned to the RMS.

The orientation shown in Figure 3.9 simplified the TM attachment operations but it is not optimum for LM attachment operations. Although feasible as oriented, the LM attachment operations could be simplified by a SOC reorientation by means of a DM rotational capability or by an undock/redock operation.

Baseline Scenario - Flight No. 7

Payload: Construction Facility (CF)

Procedures for the attachment of the construction facility to the SOC were not investigated as part of the SOC assembly task. It is appropriate, however, to recognize its place in the SOC assembly sequence. Since the CF is normally berthed to the end port of SM-2 and that port must necessarily be utilized during the assembly of SOC, it can be assembled last without affecting the operational status of the SOC.

3.2.2 SOC Assembly - Alternative Scenario No. 1

An important feature of the SOC design is that it will provide dual access to and permit shirtsleeve transfer between all pressurized habitability modules. One of the elements in this important feature is the TM. It must be assembled to the SOC before it can be declared habitable. Consequently, the sooner the TM is attached to the SOC, the sooner it can be declared operational. In Alternate Scenario No. 1 the sensitivity of the SOC assembly sequence to an earlier attachment of the TM was investigated. Compared to the baseline scenario, Alternative Scenario No. 1, advanced the assembly of the TM and LM to Flight 5 rather than Flight 6. By this sequence, the SOC can be declared operational after Flight 5, i.e., without awaiting the launch and assembly of the RCM and the SAM. Consequently, Flights 1, 2, 3, 4 and 6 are similar to Flights 1, 2, 3, 4, and 5 respectively of the baseline scenario and will not be discussed further. The entire sequence is shown in Table 3.3 and illustrated in Figure 3.10.

Alternative Scenario No. 1 - Flight No. 5

Payload: LM and TM

Equipment: PIDA, RMS, HPA and DM

Since the RCM is unavailable at this time to perform the TM attachment operations and the RMS reach is inadequate, it was found necessary to add another assembly aid - the holding and positioning aid (HPA). The attachment operations begin by docking/ berthing the SOC (SM-1 port) to the HPA. In this position, the LM and the first port of the TM are attached to the SOC. An undocking/ redocking operation follows to interface the SOC (SM-2 Port) to the HPA. In this orientation, the second TM port is attached to the SOC. A second undocking/redocking operation follows to interface the

TABLE 3.3 SOC ASSEMBLY - ALTERNATIVE SCENARIO 1 BUILDUP SEQUENCE

MODULE		CONFIGURATION 1	2	3	4	5	6	7
SERVICE MODULE 1	(SM-1)	X	X	X	X	X	X	X
SERVICE MODULE 2	(SM-2)		X	X	X	X	X	X
HABITATION MODULE 1	(HM-1)			X	X	X	X	X
HABITATION MODULE 2	(HM-2)				X	X	X	X
LOGISTICS MODULE	(LM)					X	X	X
TUNNEL	(TM)					X	X	X
STAGE ASSEMBLY	(SAM)						X	X
RMS/CONTROL	(R/CM)						X	X
CONSTRUCTION FACILITY	(CF)							X

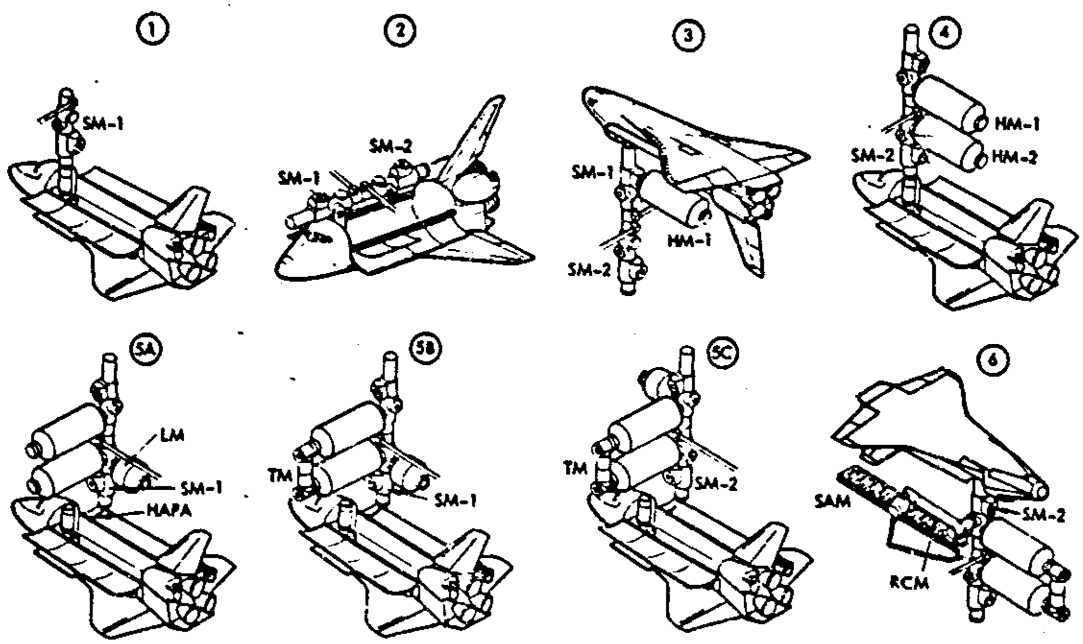

FIGURE 3.10 SOC ASSEMBLY - ALTERNATIVE SCENARIO 1

SOC to the DM. The crew can, then, transfer to the SOC after it is declared operational.

The use of a holding and positioning aid is only one alternative to circumvent the limited reach capabilities of the RMS. Others that may be considered are (a) berth the TM to the DM as the initial step, then pull the SOC with the RMS and berth it to the TM; (b) the use of a second RMS as a handling and positioning aid; and (c) resequencing the flights so that the RCM will be available to aid in the attachment of the TM to the SOC.

3.2.3 SOC Assembly Alternative Scenario No. 2

One of the noted options in the assembly of SOC was to advance the attachment of the RCM to the earliest flight possible in order to utilize its capability in a major assembly role. Alternative Scenario No. 2 presents such an option where the RCM was assigned to Flight No. 3. Consequently, its arm can be utilized for the attachment of HM-1 and HM-2. In Alternative No. 2, Flights 1, 2, 3, and 6 are similar to Flights 1, 2, 5 and 6, respectively, of the baseline and will not be discussed. The sequence of Alternative Scenario No. 2 is listed in Table 3.4 and illustrated in Figure 3.11.

Alternative Scenario No. 2 - Flight Nos. 4 and 5

Payload:	HM-2 (Flight No. 4) HM-1 (Flight No. 5)
Equipment:	PIDA, RMS, DM & RCM (both flights)

Model simulations depicted clearly the adequate reach of the RCM arm to attach both HMs. It is interesting to note that the orbiter docks to SM-1 for both flights in contrast to the baseline where the orbiter needed to dock at SM-1 and SM-2 to accommodate the RMS reach. The habitation module designated as HM-2 is attached prior to HM-1. If the sequence was similar to the baseline, then, the difficulty of maneuvering HM-2 around an attacned HM-1 would be introduced. It should be noted that the designation of HM-1 and HM-2, and similarly SM-1 and SM-2, are so by definition and does not indicate functional differences.

3.2.4 Evolutionary Buildup Concepts

The development of the assembly procedures discussed previously was premised on the acquisition of a fullup SOC as soon as possible. Alternative Scenario No. 1 offered an operational capability at the end of Flight No. 5 rather than Flight No. 6 as was the case for the baseline scenario and Alternative Scenario No. 2. These options are non-evolutionary. To develop the following two evolutionary assembly concepts, design features were added to the SOC to provide a high degree of mission flexibility. The desirability of such an approach would be further enhanced by a buildup sequence that permits the earliest operational capability and minimizes SOC front end costs. Toward that end, Evolutionary Scenarios 1 and 2 are presented.

TABLE 3.4 SOC ASSEMBLY - ALTERNATIVE SCENARIO 2 BUILDUP SEQUENCE

MODULE		CONFIGURATION 1	2	3	4	5	6	7
SERVICE MODULE 1	(SM-1)	X	X	X	X	X	X	X
SERVICE MODULE 2	(SM-2)		X	X	X	X	X	X
STAGE ASSEMBLY	(SAM)			X	X	X	X	X
RMS/CONTROL	(R/CM)			X	X	X	X	X
HABITATION MODULE 2	(HM-2)				X	X	X	X
HABITATION MODULE 1	(HM-1)					X	X	X
TUNNEL	(TM)						X	X
LOGISTICS MODULE	(LM)						X	X
CONSTRUCTION FACILITY	(CF)							X

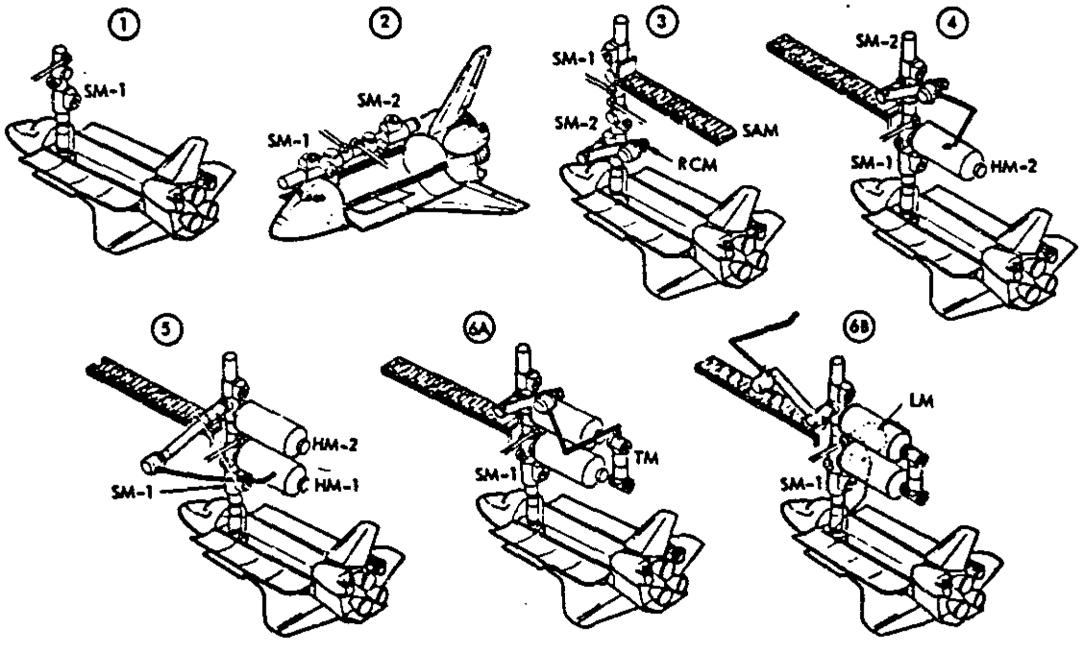

FIGURE 3.11 SOC ASSEMBLY - ALTERNATIVE SCENARIO 2

3.2.5 SOC Assembly - Evolutionary Scenario No. 1

The duration of ERNO Spacelab missions are presently limited to the orbital duration of the orbiter. To extend those missions, more power and a larger habitat capacity than those provided by the orbiter are desirable. As can be seen in the buildup sequence listed in Table 3.5 and illustrated in Figure 3.12, Evolutionary 1 concept can provide that extended mission capability as a Shuttle-tended base with one service module (SM-1) and one habitation module (HM-1). The addition of a mission module (MM), HM-2, TM and LM can provide a permanent manned base for scientific research. The third phase of this evolutionary approach adds SM-2, SAM, RCM, and the construction facility for a fully assembled SOC. The MM is an addition to the presently planned SOC modules. It is conceived as a special module or as a Spacelab with an adapter to accommodate the geometry of SOC. Flight 1 and 7 of Evolutionary 1 are the same as Flights 1 and 5 respectively of the baseline and will not be discussed.

Evolutionary Scenario No. 1 - Flight No. 2

Payload: HM-1

Equipment: PIDA, RMS and DM

A Shuttle-tended base can be declared operational at the end of this flight. The solar array of SM-1 provides the needed power capacity and HM-1 provides the needed habitation capacity. All assembly operations are within the reach capability of the RMS and nothing significantly different distinguishes this flight from the ones previously discussed.

Evolutionary Scenario No. 1 - Flight No. 3

Payload: MM

Equipment: RMS and DM

The MM was assumed to have the same envelope as a service module without its solar array, i.e., 2.5M (100 inches) by 11.73M (462 inches). The use of the Spacelab and an adapter instead of a special MM was investigated as illustrated in Figure 3.13. It revealed that a side port on the adapter can be made compatible with the subsequent HM-2 attachment and the TM attachment connecting HM-1 and HM-2. The attachment of the MM (or Spacelab/adapter combination) does not require any new assembly equipment or require any unusual maneuvers.

Evolutionary Scenario No. 1 - FLight No. 4

Payload: HM-2

Equipment: PIDA, RMS and DM

With the PIDA deploying HM-2 from the payload bay to the starboard side of the bay, the RMS maneuvering requirements are greatly minimized for

TABLE 3.5 SOC ASSEMBLY - EVOLUTIONARY SCENARIO 1 BUILDUP SEQUENCE

EVOLUTION			MODULE	CONFIGURATION 1	2	3	4	5	6	7	8
FULLY ASSEMBLED SOC	PERMANENT MANNED BASE	SHUTTLE TENDED BASE	SM-1	X	X	X	X	X	X	X	X
			HM-1		X	X	X	X	X	X	X
			MM			X	X	X	X	X	X
			HM-2				X	X	X	X	X
			TM					X	X	X	X
			LM					X	X	X	X
			SM-2						X	X	X
			SAM							X	X
			R/CM							X	X
			CF								X

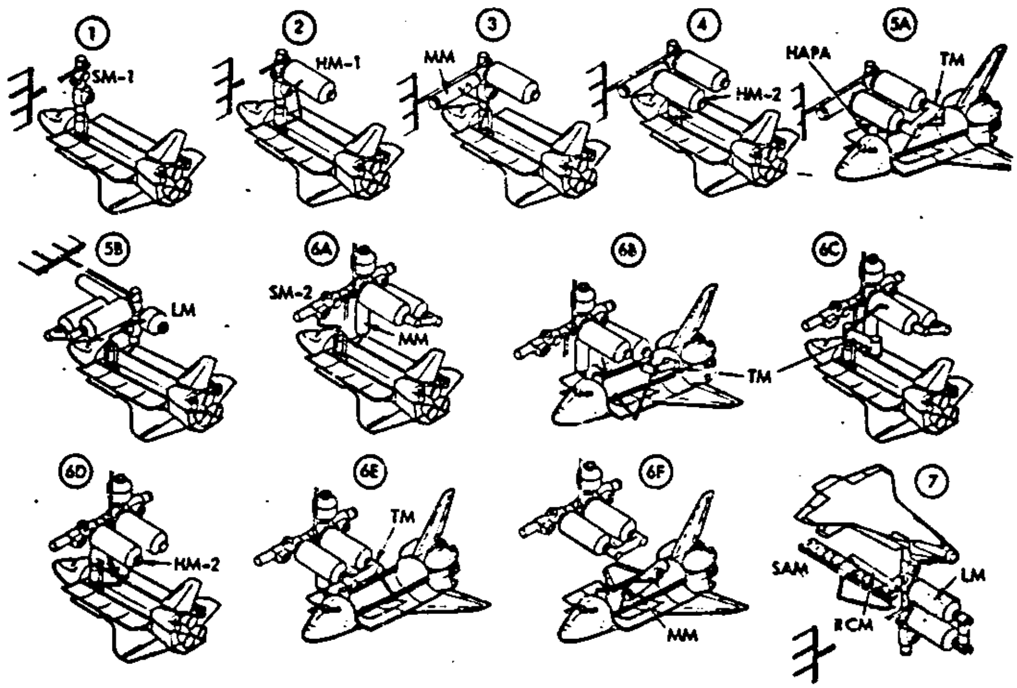

FIGURE 3.12 SOC ASSEMBLY - EVOLUTIONARY SCENARIO 1

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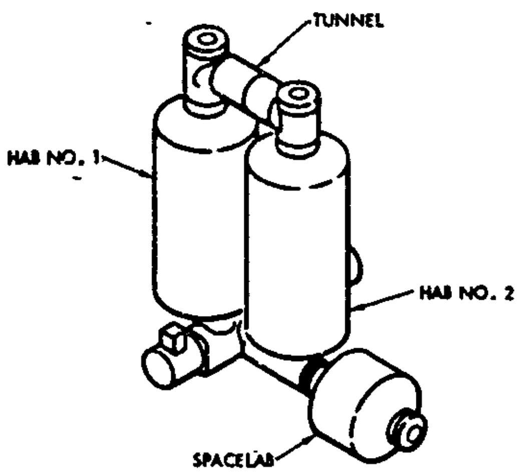

FIGURE 3.13 USE OF SPACELAB AS A MISSION MODULE

transfering HM-2 to its berthing port on the MM. As can be seen in Figure 3.12, the RMS reach is quite adequate to perform that operation.

Evolutionary Scenario No. 1 - Flight No. 5

Payload: TM and LM

Equipment: PIDA, RMS, DM and HPA

The use of a HPA for this flight is necessary for the RMS to implement the TM attachment. The specific orientation selection for the initial portion of the flight along with the HPA makes the attachment operations of both ports of the TM within the useful reach of the RMS. This observation also applies to the second portion of the flight, the attachment of the LM. One of the design features that needs to be added to the SOC to capitalize on an evolutionary buildup sequence is apparent in this flight. The relationship of the two HMs to each other is not the same as in the fullup SOC. Consequently, berthing orientations of both HMs relative to the TM are different. To overcome this peculiarity, a rotational capability on one TM port, the one that interfaces with HM-1, is required.

Since a permanently manned base can be declared operational at the end of this flight, an undocking/redocking (at the DM) operation is required to allow crew transfer into the SOC.

Evolutionary Scenario No. 1 - Flight No. 6

Payload: SM-2

Equipment: PIDA, RMS, HPA and DM

A considerable number of operations need to be performed on this flight and, for that reason, it is complex. Assuming a permanent manned base exists prior to this flight, then the initial operation is to dock the SOC to the DM and allow the SOC crew to transfer to the orbiter. Safety precautions will probably dictate that the SOC be unmanned during these attachment operations. Consequently, the initial interface port must provide manned access. If the end port of the MM does not provide that capability, then docking with the end port of SM-1 would be required. Subsequently, the SOC needs to be undocked, and redocked at the HPA/MM interface to perform the various assembly operations. The HPA/MM interface along with the selected orientation will place a series of assembly operations within the RMS reach capability. In Flight 6A, see Figure 3.12, the initial assembly operation is to berth SM-2 to SM-1. Then, subsequent to a 90° SOC rotation maneuver by the HPA, the RMS will grapple one end of the TM and a command is given to detach both its ports simultaneously from the HMs (Flight 6B in Figure 3.12). Once the TM detachment is complete, it needs to be stowed briefly for subsequent reattachment. A convenient position for the TM stowage is the DM, (Flight 6C in Figure 3.12). Simulating that operation with the 1/48 scale model indicated that the RMS articulation limits will be severely taxed and, similarly, during the reberthing of the TM to the SOC. For these reasons, grapple fixture locations on the TM will play a major role in the acceptability of this approach.

With the TM temporarily stowed, the next operation is to detach HM-2/MM interface with the RMS grappling HM-2, transfer it to its final berthing interface with SM-2 and perform the berthing. The RMS reach capabilities are adequate to perform this operation, Figure 3.12, Flight 6D.

It is necessary at this time to reattach the TM to the HMs. Again, the RMS must grapple the TM, detach it from the DM, transfer and orient it to its new berthing position and berth it to the HM-1 port. It should be noted that, with the SOC in its present orientation, (Flight 6E Figure 3.12), the RMS reach to the HM-1 port is marginal and inadequate for the HM-2 port. It will be necessary to rotate and tilt the SOC on the HPA to bring both HM ports within a workable reach of the RMS.

Once the TM is berthed to the HMs, an integrated systems checkout could be initiated. However, the present orientation of the SOC will prevent deployment of SM-2 solar array. Consequently, a reorientation on the HPA is required or the SOC/orbiter must be completely undocked in order to allow the deployment of the solar array. If reoriented, an undocking/ redocking to the DM would also be required. Return of the MM to earth is optional.

3.2.6 SOC Assembly - Evolutionary Scenario No. 2

In generating the buildup sequence of Evolutionary Scenario No. 1, all missions were assumed to be manned. In comparison, the buildup sequence of the Evolutionary Scenario No. 2 provides the added flexibility of employing SM-1 and the MM as an unmanned LEO platform by exchanging Flights 2 and 3 of evolutionary Scenario No. 1. That is, the MM will be the payload in Flight 2 and the HM-1 will be the payload in Flight 3. If the MM is assumed to be a Spacelab, the SM-1 will function as a surrogate orbiter payload bay which provides the necessary services to the Spacelab. With a revisit capability, the unmanned LEO platform will be the prelude to a fullup SOC. Flights 1, 4, 5, 6 and 7 of Evolutionary 2 are similar to Flights 1, 4, 5, 6 and 7 of Evolutionary 1, respectively, and will, therefore, not be discussed. Flights 2 and 3 and their operations are not significantly different from Flight 3 and 2 of Evolutionary Scenario No. 1. The buildup sequence of Evolutionary Scenario No. 2 is listed in Table 3.6 and illustrated in Figure 3.14.

3.2.7 SOC Assembly Conclusions

In investigating the indicating SOC assembly scenarios, the major impacts on the orbiter and SOC designs become apparent. In addition to the standard RMS and the payload bay lights, the orbiter must provide the PIDA and HPA. These pieces of construction equipment along with few delta provisions on the SOC can accommodate all possible buildup scenarios. For the SM, an interim attitude stabilization capability is required for safe revisits. An independent environmental control and life support system (ECLSS) for each of the habitation modules and a clocking capability on the TM adapter are also required to accommodate maximum planning flexibility. Both orbiter and SOC provisions are illustrated in Figure 3.15.

It is interesting to note the significance of the timing of the RCM installation to the SOC buildup. The RCM with its manipulator arm is a very feasible tool which can aid effectively in the assembly of SOC. This conclusion was verified by the 1/48 scale models. By simulating the SOC buildup with the 1/48 scale model, a high-confidence technique is proposed for the assembly of the SOC modules.

3.3 LIGHTING AND TELEVISION FOR SOC ASSEMBLY

3.3.1 Introduction

A significant area of consideration in the accomplishment of SOC assembly operations is the support equipment required to assure adequate vision. Such equipment generally can be categorized as lights, television cameras, optical alignment aids and targets. Additional aids may include sensors, software and guidance systems for final alignment, ranging and collision avoidance and windows for additional crew observation and control stations. To some degree, procedural and schedule considerations may also be appropriate, since there are frequent changes from light to dark and the reverse during flight at low earth orbit. Finally, there are considerations

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TABLE 3.6 SOC ASSEMBLY - EVOLUTIONARY SCENARIO 2 BUILDUP SEQUENCE

EVOLUTION			MODULE	CONFIGURATION 1	2	3	4	5	6	7	8
	SHUTTLE TENDED BASE	LEO PLATFORM	SM-1	X	X	X	X	X	X	X	X
			MM		X	X	X	X	X	X	X
			HM-1			X	X	X	X	X	X
PERMANENT MANNED BASE			HM-2				X	X	X	X	X
			TM					X	X	X	X
			LM					X	X	X	X
FULLY ASSEMBLED SOC			SM-2						X	X	X
			SAM							X	X
			R/CM							X	X
			CF								X

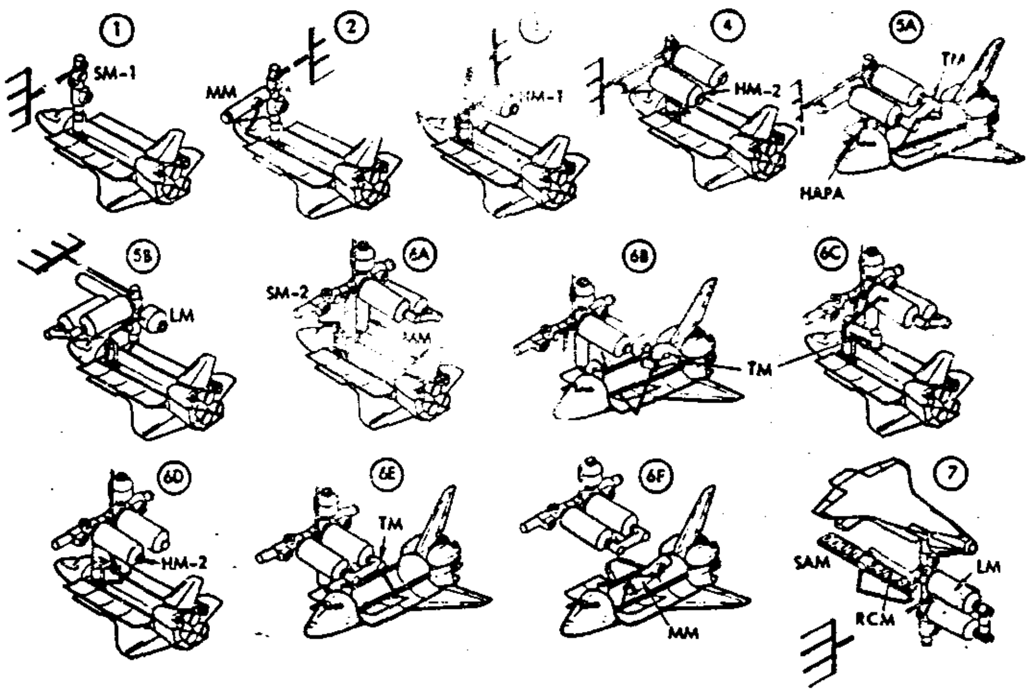

FIGURE 3.14 SOC ASSEMBLY - EVOLUTIONARY SCENARIO 2

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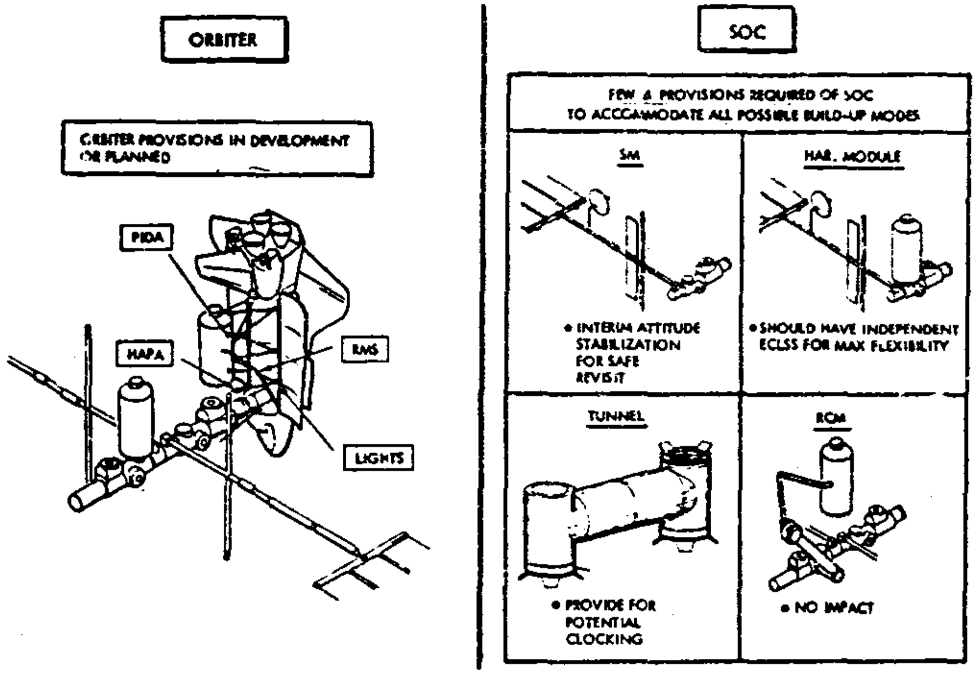

FIGURE 3.15 PROVISIONS FOR SOC ASSEMBLY

of SOC and orbiter altitude involving direct sunlight and shadow, as well as reflected light from earth, orbiter and SOC.

For purposes of the preliminary analyses performed in this study, several simplifying assumptions were made, and a generally conservative approach was taken. However, the overall concern was to keep the system cost and power requirements within reasonable bounds, and to assure adequate flexibility to adapt to future needs and possible relaxations of requirements which could simplify the systems and minimize development costs.

3.3.2 Objectives and Issues for Lighting and TV Analysis

The primary objective of the study was to develop a concept for the type of equipment needed and to determine typical locations of equipment to assist the assembly operations. The specific issues of concern included questions of whether and how much equipment should be located on the SOC or on the orbiter, whether different buildup sequences would lead to significantly different provisions, and whether or not the systems developed for SOC assembly could also be later used in SOC operations.

3.3.3 Function Analysis

Table 3.7 outlines the general functions or types of tasks to be performed in the left column, and indicates corresponding subdivisions of operating regime of particular concern on the right column. These functions and operating regimes were analyzed for each flight of five different assembly sequences to determine those aspects which required unique solutions and to seek approaches common to all. Equipment concepts were then developed which appeared to satisfy all requirements. Examples of the critical and typical concerns are presented later.

3.3.4 Assumptions, Requirements and Guidelines

The major initial assumptions, requirements and general guidelines for the study are listed in Table 3.8 below. These were derived from previous experience gained during IR&D studies and Contract NAS9-15718 "System Analysis of Space Construction", which was recently completed. The majority of supporting analyses and philosophy were reported in References 2 and 3. In such studies it was shown that lighting conditions for vision of space construction above the orbiter, during low earth orbit operations, would be generally quite good on the daylight side of orbit. However, there are possible critical conditions, such as final alignment and berthing, in which significant shadowing or exclusion of adequate light could be expected. Also, it was noted that the dark side of orbit was sufficiently long, and occurred often enough, to constitute a significant impact on productivity. Therefore, lighting to support night side operations is generally desirable to avoid work interruptions, and to minimize contingency problems. With such a philosophy, the use of daylight is normal for general transport operations, and the use of lighting is either a "fail-operational" backup or

TABLE 3.7 SOC ASSEMBLY FUNCTIONS REQUIRING LIGHTING AND TV CONSIDERATIONS

GENERAL FUNCTION	PHASE OR OPERATION REGIME	
• GRASP MODULES	• ORBITER CARGO BAY • TRANSFER—RMS TO R/CM	• SOC TO ORBITER • MODULE ON SOC
• TRANSPORT MODULES	• NEAR CARGO BAY • INTERMEDIATE DISTANCES • ORBITER TO SOC	• NEAR SOC • COLLISION AVOIDANCE • ORIENTATION
• BERTH MODULES TO SOC	• ALIGN TARGET AND SIGHT	• CLOSE AND CONTACT
• CONTINGENCIES DURING ASSEMBLY	• EVA INSPECTION • TV INSPECTION	• REMOVE/REPLACE/ RESTOW
• DOCK ORBITER TO SOC AND CAST OFF	• RENDEZVOUS • STATIONKEEP • CLOSE APPROACH	• ALIGN • CLOSE AND CONTACT • DEPART

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a normal operational mode, depending on urgency and power/energy considerations. Finally, the dependence on daylight alone is a backup mode if the lighting system fails or is deemed inadequate for night side operations in a particular flight.

The remainder of the assumptions and guidelines relate to cost minimization and recognition of potential problems of excess power demand from over-generous lighting provisions. Since one guideline may be in conflict with another, a decision needs to be made in several cases as to which will be given greatest weight in making a recommendation.

TABLE 3.8 ASSUMPTIONS, REQUIREMENTS AND GUIDELINES FOR SOC ASSEMBLY, LIGHTING AND TELEVISION ANALYSES

ASSUMPTIONS	RATIONALE & IMPLICATIONS
• ASSUME NIGHTSIDE OPERATIONS TO BE ACCOMMODATED	• DESIRABLE TO ENHANCE PRODUCTIVITY OF MISSION, PROVIDE FOR CONTINGENCIES
• ASSUME ATTITUDE OF SOC & ORBITER NOT CONTROLLED FOR OPTIMUM LIGHTING	• MINIMIZE FUEL, STRESSES DUE TO GRAVITY GRADIENTS & RCS FIRINGS, AVOID WORK INTERRUPTIONS
• ASSUME CREW USES DAYLIGHT FILTERS, DARK GLASSES, & PARTIAL DARK ADAPATATION DURING REST PERIODS	• MINIMIZE LIGHTING POWER REQUIREMENTS (MINIMAL EFFECT ON PRELIMINARY ANALYSES)
REQUIREMENTS	
• PROVIDE ADEQUATE ILLUMINATION FOR DIRECT VISION WHERE FEASIBLE	• DESIRABLE FOR CREW OPERATIONS, ACCURACY, PRODUCTIVITY
• PROVIDE ADEQUATE ILLUMINATION FOR TV WHERE DIRECT VISION NOT FEASIBLE OR ADEQUATE	• TV GIVES GOOD UNDERSTANDING OF CLEARANCES, ATTITUDES; LOW-LIGHT-LEVEL TV NEEDS LESS LIGHT THAN HUMAN VISION IN MANY CASES
GUIDELINES	**RATIONALE & IMPLICATIONS**
• PROVIDE LOCALIZED LIGHT SOURCES RATHER THAN GENERAL FLOOD-LIGHTING	• MINIMIZE POWER DEMAND FOR BOTH ORBITER & SOC; BASED ON PREVIOUS ANALYSES
• UTILIZE SOC POWER FOR LIGHTING WHERE FEASIBLE	• MINIMIZE ORBITER POWER & ENERGY REQUIREMENTS, ORBITER PAYLOAD WEIGHT
• MINIMIZE ORBITER DESIGN IMPACTS	• MINIMIZE ORBITER TURNAROUND FOR NON-DEDICATED ORBITER
- MINIMIZE ELECTRICAL WIRING AND CONNECTORS	• MINIMIZE P/L INTEGRATION COSTS & WEIGHT
- MINIMIZE SPECIAL TV & LIGHTING FIXTURES	• MINIMIZE WEIGHT, POTENTIAL P/L ENVELOPE INCURSIONS & INTEGRATION COSTS
• MINIMIZE PENETRATION OF PRESSURE VESSELS	• RELIABILITY & CREW SAFETY CONSIDERATIONS
• STANDARDIZE METHODS & EQUIPMENT WHERE FEASIBLE	• MINIMIZE COSTS OF TRAINING & NEW SPECIAL EQUIPMENT DESIGN & INVENTORY
• PROVIDE INDIRECT DIFFUSE LIGHT	• AVOID GLARE AND TV "BLOOMING," GAIN SETTING PROBLEMS

3.3.5 Analysis

The lighting and television analysis was primarily based on review of the geometry of the orbiter and SOC components at the critical stage of final alignment and berthing, as described in Section 3.2. Each flight was reviewed for each of five build sequences. Similarities and differences were noted, and requirements for typical and unique situations were identified. Among the driver activities were the following:

1. Initial module setup and checkout--berthing SOC module to orbiter docking port using RMS.

2. Initial rendezvous and docking of orbiter to SOC module and considerations for future docking.

3. Initial module-to-module berthing, using orbiter RMS (close to cargo bay).

4. Module-to-module berthing using the RMS, at considerable distance from the orbiter.

5. RCM used for module-to-module berthing.

6. Use of Handling and Positioning Aid (HPA) to hold a specified relationship between orbiter and SOC.

In addition, certain differences were noted in module configuration (one berthing port or two, as in the Tunnel Module, and the requirement for multiple reuse, as for the Logistics Module. Each of these distinct differences potentially introduced a new factor for consideration in the lighting and television requirements. However, potential methods for handling one problem also were found useful in handling others, and were considered as candidate methods until the full analysis was completed. Then, the totality of options was reviewed and selections made in accordance with the guidelines described in Section 3.3.4.

A small number of examples are discussed herein to illustrate some of the important considerations and to show recommended methods for dealing with them. The first example used here involves Flight No. 3 of the baseline assembly sequence, shown in Figure 3.16.

A significantly illustrated point is that the extended docking module, located just aft of the crew cabin, blocks off most of the light from the standard fixed lamp between the aft facing windows. Furthermore, it also obstructs much of the crew's vision into the cargo bay and above it. Therefore, some replacement capability is required to see the process of transporting the habitability module from the cargo bay up to its designated port for berthing on the SOC. Figure 3.17 illustrates a proposed method. Two movable TV cameras and associated lights are mounted on the aft, exterior sides of the upper portion of the Docking Module to supplement the obstructed standard lighting and vision. The standard TV camera location on the forward bulkhead is not used since it could have only limited aft/upward vision. Figure 3.17 also recommends a method for alignment of the first SOC module to the Docking Module, consisting of a TV camera and lights inside the docking port. This approach was selected as most compatible with the standard RMS manipulator control station operation (which has no crewman optical alignment sight suitable for use with an external target). The same approach is recommended for berthing the module HM-1 (and several other modules) to the SOC, since direct vision from the orbiter upper windows and light from the top of the orbiter cabin are both obscured by the previously assembled portions of the SOC docked to the orbiter. Vision of the final berthing activity is also limited from the TV camera on the aft bulkhead of the cargo bay. While the RMS elbow camera could aid in determining range for the berthing operation, it cannot do well as an alignment aid. Figure 3.2 further illustrates the basic concept for this kit, which involves a TV camera and a set of lights on the SOC port (or orbiter port) and a target on the upcoming module port. This system assures adequate light at all times, even as the ports and drawn together for final latchup.

Initial conception of this approach was accomplished in connection with a layout drawing study to develop assembly alignment aids. The results, shown in Drawing 42690-009, illustrate a berthing port concept incorporating a TV camera on one port and an indirectly-illuminated target on the mating port. However, this concept would require supplying power to both ports, directly (by hard line) or by use of batteries and switching functions operated by hard line on RF. Figure 3.2 features a concept in which both active elements (lights and TV camera) are on just one port while the other contains only the passive target. This concept also aids the TV camera vision by illuminating a larger area of the module approaching it, probably from a greater distance away.

Careful consideration was given to which side of the berthing port should contain the active or the passive elements. It was concluded that the active elements (TV and lights) should be on the side of the already-assembled SOC in every case. This assures capability of hard line connections from activated vehicle elements and is believed to be compatible with other crew control display relationships as obtained from remotely located TV (and outside window viewing where possible). Tests performed on the Manipulator Development Facility at JSC seem to confirm that this concept can be used by trained operators.

Figure 3.16 also indicates proposed methods for aiding rendezvous, station keeping, approach and docking of the orbiter to the SOC. To serve as a clear indication of docking port locations, four small, colored marker lights are recommended, such that the docking port location can be observed from any reasonable approach direction. In addition, low-wattage marker lights at major extremities of the assembled SOC are also recommended, to aid in collision avoidance. For final alignment and docking, it is recommended that the standard crew optical alignment sight and exterior

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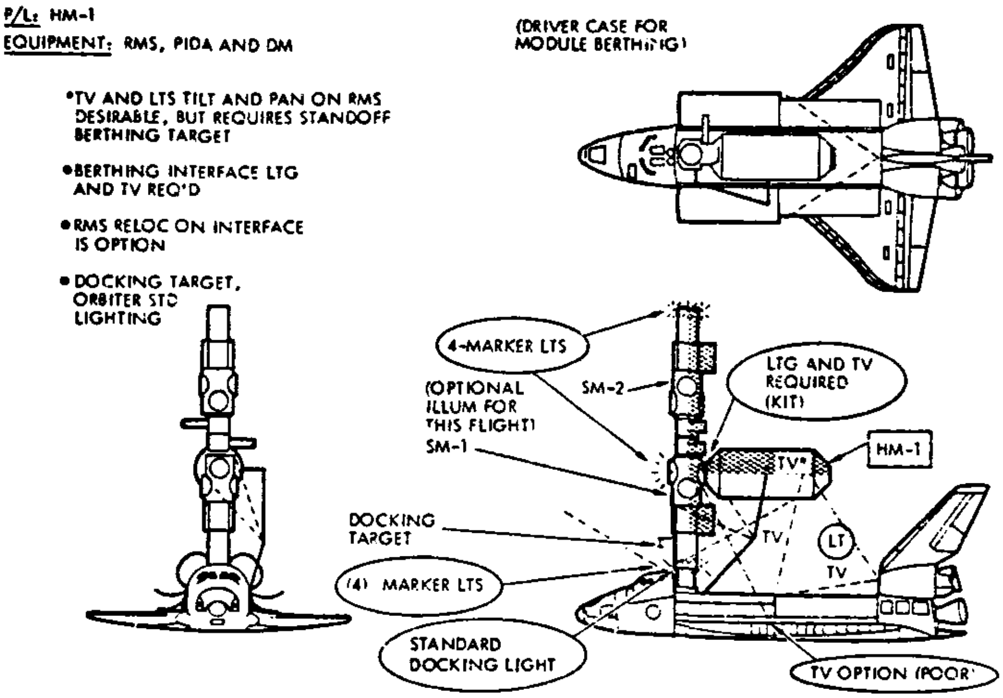

FIGURE 3.16 LIGHTING AND TV CONSIDERATIONS FOR BASELINE FLIGHT 3

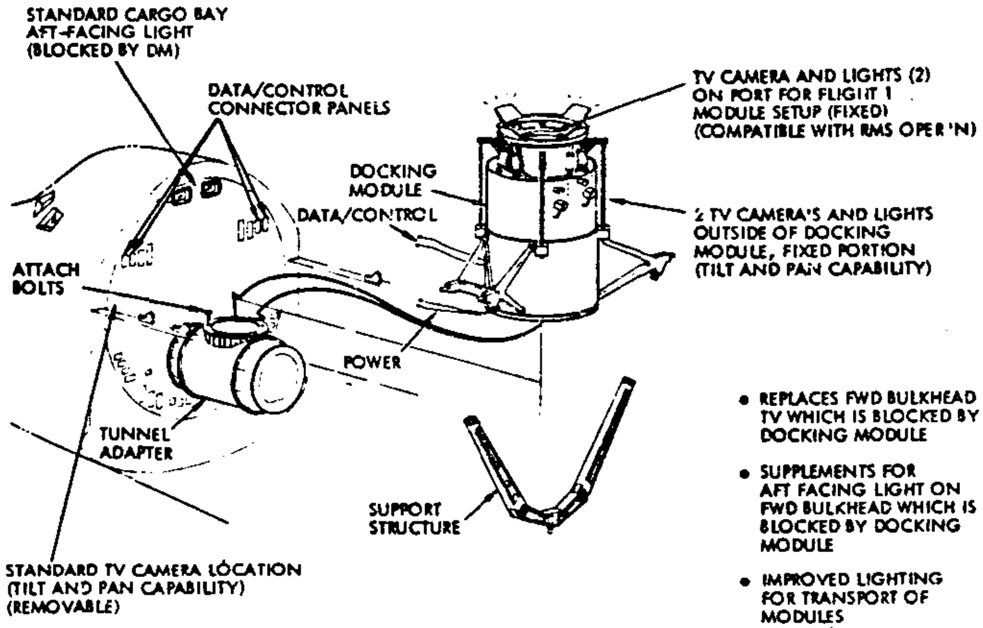

FIGURE 3.17 LIGHTING AND TV RECOMMENDED FOR DOCKING MODULE

docking light be used with externally mounted targets on the SOC. This concept is compatible with the standard orbiter aft flight deck control and display layout concept. As an alternative procedure (not recommended in this study) it may be found desirable to use the orbiter RMS to berth the orbiter to the SOC. In this case, the aforementioned internal TV camera and target concept can be used by the manipulator operator.

Another concern of lighting for SOC assembly is the avoidance of collisions between moving modules and the SOC or the orbiter (especially the tail) during transport from cargo bay to SOC berthing. To aid this gross transport phase it appears that some additional, steerable lighting is required near the cargo bay upper surfaces and considerably further aft than the docking module. A natural location for such lighting appears to be the aft bulkhead. A lamp on the elbow of the RMS wrist could also be beneficial, but is judged as more difficult to stow and integrate with the orbiter. Another alternative would be to mount such lamp or lamps on PIDA arms, which may be required to handle the large SOC modules. Lights mounted on the outer edges of the payload bay doors or sill could also be utilized if space can be found which does not interfere with payload envelope space or the door latches. Another possible aid could be lights mounted on the airlock surfaces so as to illuminate the berthing ports for modules HM-1 and HM-2. These would have some potential value for future EVA operations. However, since lamps within the ring of the berthing port seem to be the most generally usable, standard concept for most SOC assembly berthing operations, the use of other externally mounted lights on the SOC was not recommended for this study.

Another strongly considered alternative was providing a tilt and pan capability for the RMS wrist TV camera and light. Such lighting and vision capability could be useful in the transport phase and could be used in some cases for final alignment and closure guidance during berthing. However, it would pose a requirement for several different types of mountings for external targets (often with quite large support structures) on the SOC and could significantly restrict options for location of grapple points on the SOC modules. For example, it would tend to strongly favor grapple fixtures located close to the berthing port. Such a requirement could make reach and clearance for the RMS arm more difficult, and possibly could cause difficulties from having a greater distance of the module center of mass from the point of handling. On the other hand, grappling near the berthing port could improve accuracy of the berthing control by reducing concerns from effects of small angular errors when holding long modules near their center (such as shown in Figure 3.16). While tilt and pan capability of the orbiter RMS wrist camera and light are considered desirable, this capability does not appear essential at this time to accomplish the SOC assembly. Therefore, it is not included among the final recommendations.

Flight 5 of the baseline sequence provides another example of major lighting and television concerns, primarily those dealing with the RMS Control Module (RCM). Key concerns are illustrated in Figure 3.18. This method of handling modules is used at distances where the RMS cannot reach, and concurrently the orbiter lights will tend to be relatively ineffective. Therefore, it is recommended that steerable lights be attached to the exterior of the RCM cabin, as well as the elbow and the wrist of the manipulator. Also, steerable TV cameras at the manipulation wrist and elbow are recommended for the RCM manipulator. Note that the berthing port TV camera in the concept previously described should be transmitting scenes to the RCM operator when this crew member is handling modules.

The use of a handling and positioning aid (HPA) is considered in some of the SOC assembly sequences studied (Figure 3.19). Such an aid, if used, should have a TV camera and light system on its interface with the SOC docking/berthing port, so as to assure accurate, safe final alignment when dealing with the large masses of the orbiter and SOC. These cases would again require use of a berthing target on the SOC port to which the HPA is attached (usually the docking port). In general, the use of the HPA did not appear to significantly change other needs for accurate vision and lighting of the critical berthing port interfaces. Some minor improvements and options in vision/ lighting equipment were noted on specific flights, but the foregoing major recommendations appeared valid for most cases.

Table 3.9 provides a summary of lighting and TV recommendations, according to location on the SOC. Table 3.10 is a similar summary for the orbiter and its equipment (DM, HPA). It was concluded that only minor changes are required in the standard orbiter payload provisions, and most of these are involved with removable devices. In fact, the final detail designs for the docking module are yet to be engineered, and such items as the HPA are only in preliminary study stages.

Table 3.11 provides a summary of types of hardware recommended for development for SOC assembly (and later use). The number of different items is relatively small, since similar devices can be used in many places, both in the orbiter and the SOC. In fact, the basic elements as indicated probably require no new technology development.

However, some consideration should be given to optimizing light levels and wavelengths for the berthing port and target concept, so as to minimize glare and unwanted reflections. Options include use of a mix of ultraviolet light and visible light with florescent target surfaces. Another possibility is use of reflective tape on the target, and reduced light levels. Decisions as to which method to be used would be based upon mockup evaluations and simulations. The result could influence some technology review and adaptation activity.

In addition to the geometric analyses, an estimate of peak power requirements was performed for the SOC and orbiter. The results are given in Table 3.12 below for a nearly completed SOC buildup, which includes installation of a construction fixture module, but not a service facility.

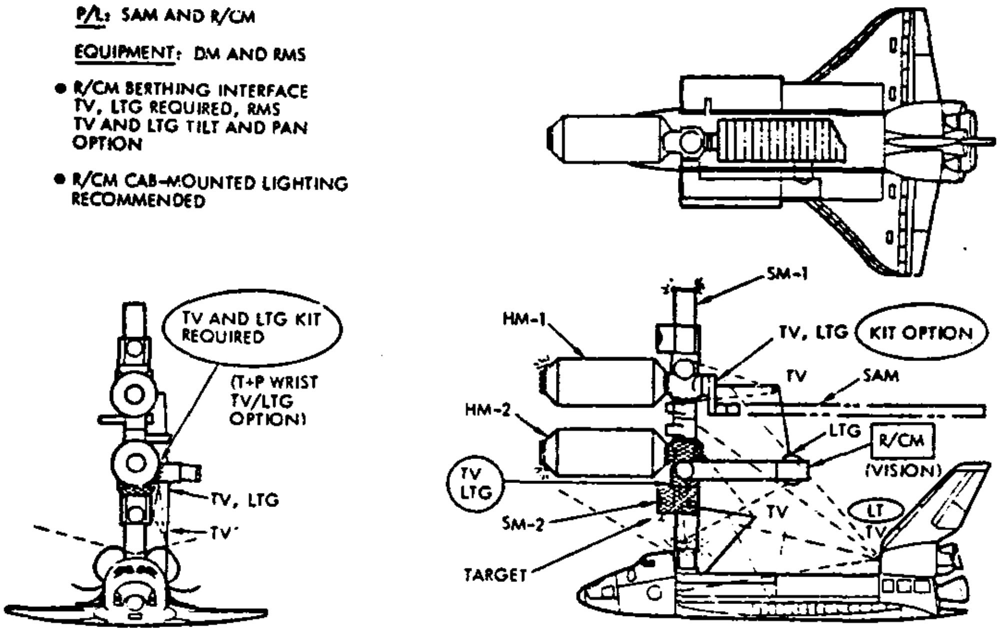

FIGURE 3.18 LIGHTING AND TV CONSIDERATIONS FOR BASELINE, FLIGHT 5

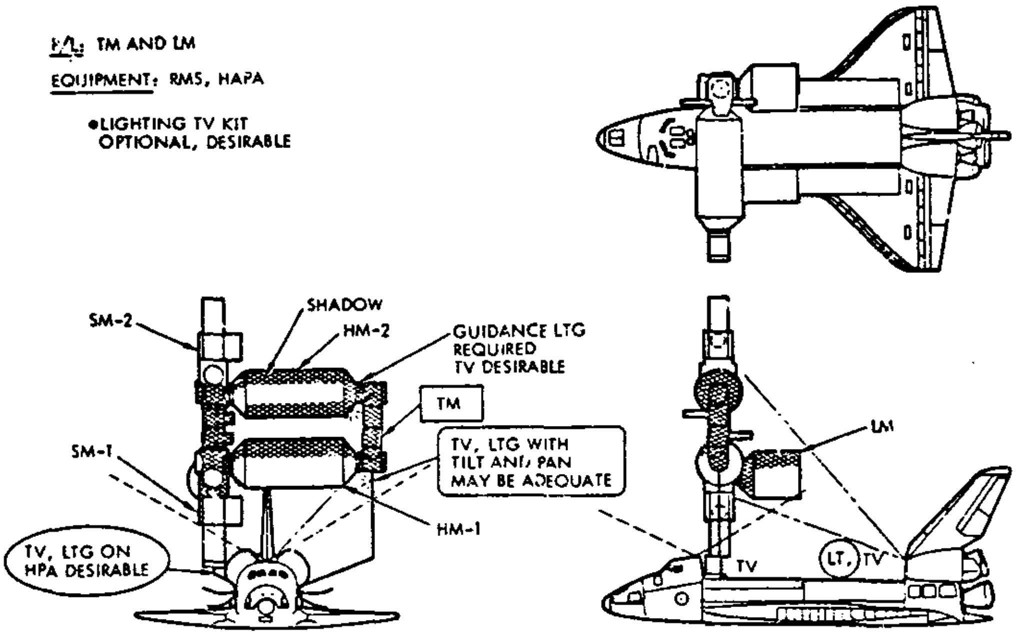

FIGURE 3.19 LIGHTING AND TV CONSIDERATIONS FOR ALTERNATIVE 1, FLIGHT 5B

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TABLE 3.9 SUMMARY - LIGHTING AND TV RECOMMENDATIONS (BY LOCATION) FOR THE SOC

LOCATION	PROVISIONS RECOMMENDED BASELINE	ALT. 1	ALT. 2	EVOL. 1	EVOL. 2	SUMMARY
SM-1/LM BERTH PORT	KIT 1	LIKE B/L	LIKE B/L	LIKE B/L	LIKE B/L	LIKE BASELINE (B/L)
HM-1/TM OR HM-2/TM BERTHING PORT (FIRST ATTACHMENT)	R/CM MANIP. WRIST TV & LT (T&P), ELBOW & AUX. TV (OR KIT 1)	RMS WR. TV & LT (T&P), ELBOW & AUX. TV (OR KIT 1)	LIKE B/L	LIKE ALT. 1	LIKE ALT. 1	DEPENDS ON BUILD PLAN. KIT 1 RECOMMENDED
SAME AS ABOVE (SECOND ATTACHMENT)	NONE	KIT 1 DESIRED	KIT 1 DESIRED	KIT 1 DESIRED	KIT 1 DESIRED	KIT 1 RECOMMENDED
HPA/SM-1 OR SM-2 (SOC SIDE)	N/A	KIT 1 (TARGET)	N/A	KIT 1 (TARGET)	KIT 1 (TARGET)	KIT 1 (IF HPA)
SM-1/SAM BERTHING PORT	R/CM TV & LTS (OR KIT 1)	LIKE B/1	LIKE B/L	LIKE B/1	LIKE B/1	LIKE B/L
SM-1/MM BERTH PORT	N/A	N/A	N/A	KIT 1	KIT 1	KIT 1
SM-1 DOCKING PORTS & EXTREMITIES	8-MARKER LTS 10-MARKER LTS	LIKE B/L	LIKE B/L	LIKE B/L	LIKE B/L	LIKE B/L
SM-2 DOCKING PORT & EXTEMITIES	8-MARKER & 10-MARKER LTS	LIKE B/L	LIKE B/L	LIKE B/L	LIKE B/L	LIKE B/L
Δ EXTREMITIES MARKER LTS	13	13	13	13	13	13
SM-1/HM-1 BERTHING PORT	2 LTS, FIXED TV FIXED (KIT 1)	LIKE B/L	NOT REQ/D	LIKE B/L	LIKE B/L	LIKE B/L
SM-2/HM-2 BERTH PORT	(KIT 1)	LIKE B/L	DESIRED	LIKE B/L (RMS OPT)	LIKE B/L (RMS OPT)	LIKE B/L
SM-2/RC/M BERTHING PORT	RMS, WR. T&P OR KIT (OPT)	LIKE B/L	LIKE B/L	LIKE B/L	LIKE B/L	RMS WR. T&P (OR KIT)
R/CM CABIN EXTERIOR	4 LTS, TILT & PAN	LIKE B/L	LIKE B/L	LIKE B/L	LIKE B/L	LIKE B/L
R/CM MANIP. WRIST	1-LT T&PAN 1-TV T&PAN	LIKE B/L	LIKE B/L	LIKE B/L	LIKE B/L	LIKE B/L
R/CM MANIP ELBOW	1-LT T&PAN 1-TV T&PAN	LIKE B/L	LIKE B/L	LIKE B/L	LIKE B/L	LIKE B/L
CONSTR. FIXTURE/ SM-2 BERTH PORT	KIT 1	KIT 1	KIT 1	KIT 1	KIT 1	KIT 1
FLIGHT SUPPORT FACILITY	← NOT ANALYZED →					

TABLE 3.10 SUMMARY - LIGHTING AND TV RECOMMENDATIONS (BY LOCATION) FOR THE ORBITER

LOCATION	PROVISIONS RECOMMENDED					SUMMARY
	BASELINE	ALT. 1	ALT. 2	EVOL. 1	EVOL. 2	
DM OUTSIDE	2-LTS, TILT & PAN 2-TV, TILT & PAN	LIKE B/L	LIKE B/L	LIKE B/L	LIKE B/L	LIKE B/L
DM PORT	2-LTS, FXD 1-TV, FXD (KIT 1)	LIKE B/L	LIKE B/L	LIKE B/L	LIKE B/L	LIKE B/L
AFT BULKHEAD OF CARGO BAY	1-LT, TILT & PAN	LIKE B/L	LIKE B/L	LIKE B/L	LIKE B/L	LIKE B/L
HPA/SM-1 OR SM-2 (REF.)	N/A	KIT 1 LTS/TV	N/A	KIT 1 LTS/TV	KIT 1 LTS/TV	KIT 1 LTS/TV (IF USED)

TABLE 3.11 SUMMARY OF LIGHTING AND TV INSTALLATION FOR SOC ASSEMBLY

TYPE OF INSTALLATION	NUMBER OF PROVISIONS RECOMMENDED					
	BASELINE	ALT. 1	ALT. 2	EVOL. 1	EVOL. 2	MAX.
SOC						
• BERTHING PORT						
- TV & 2 LIGHTS (SET)	4-8*	4-8*	2-5*	5-9*	5-9*	9*
- TARGETS (ALONE)	2	2	1	2	2	2
• R/CM CABIN LAMPS (T&P)	4	4	4	4	4	4
• R/CM MANIPULATOR						
- WRIST LT & TV (T&P)	1	1	1	1	1	1
- ELBOW LT & TV (T&P)	1	1	1	1	1	1
• MARKER LIGHTS AT EXTREMITIES (COLORED)	45	45	45	45	45	45
ORBITER						
• ORBITER DOCKING MODULE						
- EXTERIOR LT & TV (T&P)	2	2	2	2	2	2
- DOCK PORT LT & TV (KIT)	1	1	1	1	1	1
• RMS WRIST TV & LT (T&P)		1		1		NOT ESSEN.
• AFT BULKHEAD LT (T&P)	1	1	1	1	1	1
• HPA LT & TV (KIT)		1		1		(IF USED)

*SOME REUSABLE

Table 3.12

Estimated Peak Power Requirements for SOC Assembly Operations

Vehicle	Buildup Concept Baseline	Alternative 1 (HPA)
SOC	6.6 kW	6.6 kW
Orbiter	4.5 kW	6.1 kW

These results indicate the orbiter power demand for the orbiter's payload operations (which include RMS power and heat, all cargo bay lights and TV cameras) is less than the 7 kW maximum continuous power supply capability. Since the average power demand will be lower, and the total assembly operation time is probably relatively short (3-6 hours), the baseline payload-dedicated energy supply of 50 kWh will probably not be exceeded.

In the above power analysis it was assumed the SOC would supply its own needed power from a partially deployed solar array and batteries, after the first service module was placed on orbit. SOC "housekeeping" power demands were not included.

3.3.6 Lighting and Television Analysis Conclusions and Recommendations

It was concluded that only minimal impacts to the orbiter result from lighting and television requirements for SOC assembly. These impacts are no worse than the expected changes to wiring, connections, and brackets which could be expected in integration of other satellites or sortie experiments, and could include either readily-removable, clamp-on devices or permanently installed provisions for a steerable (tilt and pan) light on the aft bulkhead of the cargo bay (or other nearby site, such as the sill) and wiring to connect the docking module (and possibly HPA) TV cameras and lights to the aft cabin TV screens and switching controls.

The recommended SOC lighting provisions would also represent a partial list of the total SOC requirements for operational use. Some of the recommended TV cameras and lights for berthing ports could be removable and reusable once the modules are installed. The power demand is considered acceptable for the assembly phase of SOC lifetime, and lighting does not appear to be a major driver among total SOC systems requirements.

The reader is referred to the summary outline in Section 3.1 for an illustration of the overall lighting, television and target recommendations (Figure 3.3).

3.4 CONTROL IMPLICATIONS DURING UNTENDED OPERATIONS

Another important Shuttle-related question for SOC assembly relates to the capability of the SOC to provide a safe and stabilized target properly oriented for orbiter revisit and docking. Of principal concern here, is whether the control system elements sized for the full up SOC configuration can satisfy the widely varying configuration arrangements and partial system availabilities inherent in the modular delivery and buildup of the large SOC system. Early use of SOC configurations comprised of fewer modules in different arrangements to perform Shuttle tended space construction operations and/or OTV flight support operations could introduce requirements levels for the partial system which exceed those in a direct buildup plan. This is particularly true for cases where only one service module and its asymmetrical solar array arrangement are used in conjunction with other modules and equipment to perform early space operations. In the direct buildup case SOC based support operations would not commence until the complete SOC configuration was assembled. This could allow less demanding stabilization and control requirements to be imposed during the partial assembly phases.

Thus, Shuttle interactions with SOC attitude control during untended operations were briefly examined. The assumed system requirements, configuration and mass property variations and control system sizing analyses are presented in the following text.

3.4.1 Assumed System Requirements

The requirements used in analyzing the SOC dynamics and control characteristics, and in synthesizing the guidance, navigation, and control (GN&C) systems necessary for Shuttle interfacing, are as follows:

1. The SOC angular rates during docking must be below those specified in the preliminary docking contact conditions (0.6 deg/sec).

2. During docking and assembly operations the SOC will be required to maintain attitude within ± 0.5 degrees; however, for most other requirements a ± 5 degree attitude deadband will be sufficient since there are no precise pointing requirements imposed by space construction and flight support operations.

3. Post-docking disturbances will be damped by SOC, and attitude control of the SOC-orbiter combination will be performed by SOC during an orbiter visit. The orbiter control system will be inactivated.

4. Momentum dumping (CMG desaturation) will be performed by the SOC RCS in such a manner as to simultaneously generate orbit makeup impulse as much as is practical.

5. The SOC RCS will have three-axis angular control and +X translational thrusting capability, with thruster and propellant feed system redundancy in all functions.

6. The SOC solar panels will have two-axis gimbals for efficient solar viewing.

In addition, there are several "soft" requirements, or preferences, as follows:

1. The SOC will be located in circular 350 $\pm$ 50 KM orbits at 28.5 degree inclination.

2. The orbiter will normally dock by approaching along the SOC velocity vector (orbiter -Z axis aligned with the SOC +X axis).

3. RCS moment couples will be used in attitude control where possible to reduce interaxis coupling propellant losses (this applies primarily to roll attitude control, but should be available in the pitch and yaw axes).

3.4.2 Buildup Sequence and Configuration Options

The SOC basic elements and representative user systems (OTV, etc) are pictured in Figure 3.20. They are listed from top to bottom in essentially the order prescribed for the direct buildup sequence. An alternate evolutionary buildup sequence used in the study differs in a number of aspects; for example, it employs a logistics module after the service Module 1 has been deployed. Also, an alternate orbiter docking mode was briefly investigated to permit a more aerodynamically-balanced service module-orbiter configuration.

Both orbiter and SOC axis systems are defined in Figure 3.21. The relationship shown in the figure is indicative of the primary docking orientation of the orbiter to the SOC, referred to as "Orbiter A".

Specific configurations for the various SOC buildup stages covering an example evolutionary build plan as well as the direct buildup case are identified in Figure 3.22. While the direct sequence reaches full SOC capability in fewer assembly stages, the evolutionary sequence permits utilization of the first service module as a shuttle-tended platform for an extended period of time. If this evolutionary sequence is used, the alternate orientation described as "Orbiter B" should be considered, since it can potentially reduce aerodynamic disturbances.

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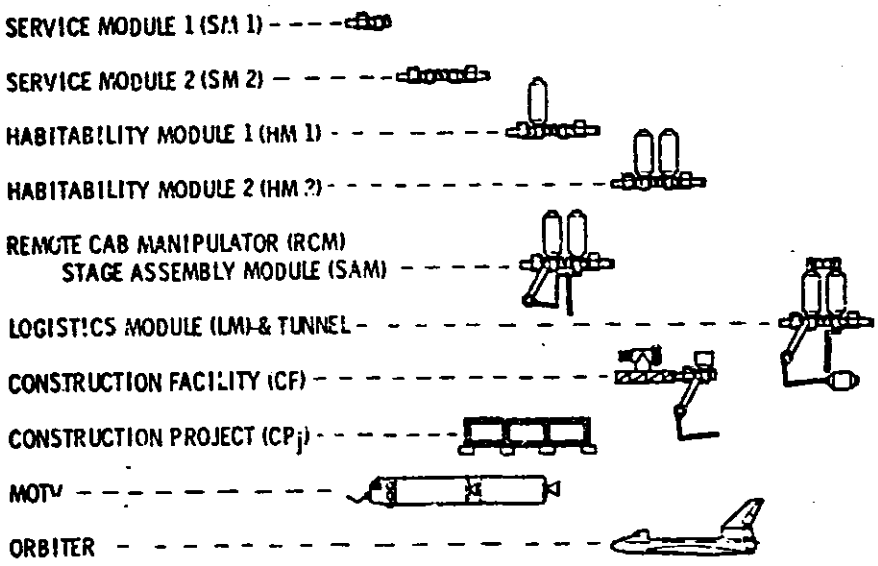

FIGURE 3.20 SOC CONFIGURATION ELEMENTS

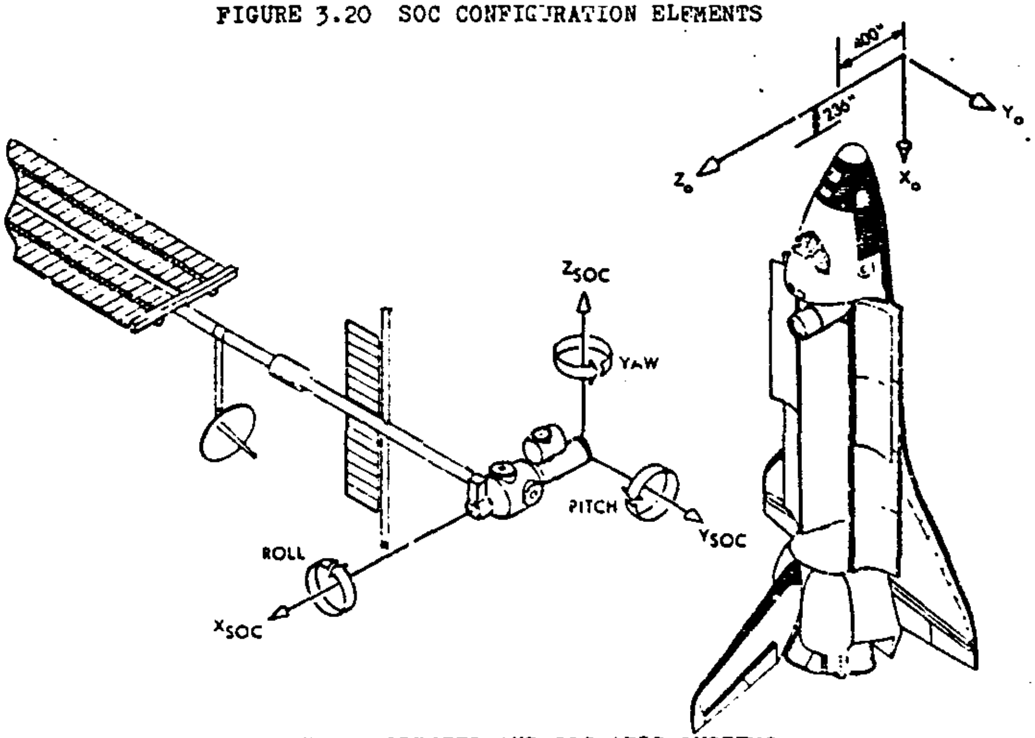

FIGURE 3.21 ORBITER AND SOC AXIS SYSTEMS

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EVOLUTIONARY (INCL 31-38) (cases 1–21) — DIRECT (INCL 1, 3, 4, 31-38) (cases 22–30)

CASE	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26	27	28	29	30
SERVICE MODULE 1	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
SERVICE MODULE 2														X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
HAB MODULE 1								X	X	X	X	X	X	X	X	X	X	X	X	X	X		X	X	X	X	X	X	X	X
HAB MODULE 2																		X	X	X	X				X	X	X	X	X	X
RMS/CONTROL																X	X			X	X						X	X	X	X
STAGE ASSEMBLY																X	X			X	X						X	X	X	X
TUNNEL MODULE																			X	X	X								X	X
LOGISTICS MODULE					X	X	X				X	X	X	X		X	X	X	X	X	X								X	X
CONST FACILITY																														
CONST PROJECT																														
OTV																														
ORBITER A	X					X			X			X			X		X		X		X	X		X		X		X		X
ORBITER B		X					X			X			X																	

SOLAR ARRAY STOWED (case 3)
SOLAR ARRAY DEPLOYED (case 4)

CASE	31	32	33	34	35	36	37	38
SM1	X	X	X	X	X	X	X	X
SM2	X	X	X	X	X	X	X	X
HM1	X	X	X	X	X	X	X	X
HM2	X	X	X	X	X	X	X	X
R/CM	X	X	X	X	X	X	X	X
SAM	X	X	X	X	X	X	X	X
TM	X	X	X	X	X	X	X	X
LM	X	X	X	X	X	X	X	X
CF	X	X			X	X	X	X
CP					X	X	X	X
OTV			X	X			X	X
ORBITER A		X		X		X		X

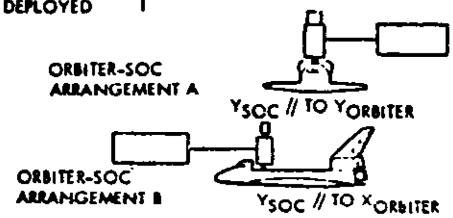

FIGURE 3.22 SOC CONFIGURATIONS OPTIONS MATRIX

TABLE 3.13 MASS PROPERTIES FOR SOC BUILDUP CONFIGURATION

CONFIG	WEIGHT (LB)	CENTER OF GRAVITY (FT)			MOMENT OF INERTIA (SLUG-FT²)					
		X	Y	Z	I_{XX}	I_{YY}	I_{ZZ}	I_{XY}	I_{YZ}	I_{ZX}
1	42,174	+23.60	-18.79	+1.30	2,358,918	253,914	2,498,368	-292,189	-60,726	+15,398
2	84,348	+38.50	0	+1.30	5,643,570	1,089,942	6,504,580	+149,706	0	0
3	122,612	+35.38	0	+8.51	6,183,076	1,710,900	6,615,180	+149,475	0	-189,140
4	160,876	+38.50	0	+12.30	6,514,602	2,198,438	6,800,384	+149,475	0	0
5	180,174	+36.69	-1.61	+10.98	6,733,029	2,414,697	7,058,329	+269,991	+98,800	+98,800
6	186,004	+36.95	-1.55	+12.29	7,004,983	2,703,438	7,077,019	+270,504	+110,569	+110,569
7	204,420	+36.84	-1.42	+7.19	9,078,361	4,770,734	7,085,763	+269,508	+64,725	+148,548
8	223,736	+38.28	+0.19	+4.79	9,859,623	5,499,388	7,446,229	+439,980	-252,966	-105,498
9	245,142	+43.69	+0.18	+3.69	10,041,413	8,269,753	10,047,094	+432,403	-251,472	-648,327

3.4.3 Mass Properties Development

Mass properties data were generated for many of the cases identified. The basic information was drawn from the JSC-supplied data given in Table 3.13 and Figure 3.23. (Note that the configuration numbers do not match the case numbers used elsewhere in this section.) In addition, the following "orbiter A" mass properties were used:

			ENGLISH UNITS		METRIC UNITS	
Weight	W	=	175718	lb	79704	kg
CG in SOC axes	X	=	- 8.33	ft	- 2.54	M
	Y	=	0		0	
	Z	=	- 39.58	ft	-12.06	M
Moments and Products of Inertia in Orbiter axes, about orbiter CG	I_{XX}	=	805,000	slug-ft^2	1,091,419	kg-M^2
	I_{YY}	=	6,112,000	slug-ft^2	8,286,650	kg-M^2
	I_{ZZ}	=	6,377,000	slug-ft^2	8,645,937	kg-M^2
	I_{XY}	=	-6,000	slug-ft^2	-8,135	kg-M^2
	I_{XZ}	=	242,000	slug-ft^2	328,104	kg-M^2
	I_{YZ}	=	-20,000	slug-ft^2	-27,116	kg-M^2

The actual mass properties calculations were performed using one function of NASTRAN, a computer program widely employed in structural dynamics analyses.

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MODULE	CONFIGURATION								
	1	2	3	4	5	6	7	8	9
SERVICE MODULE 1	X	X	X	X	X	X	X	X	X
SERVICE MODULE 2		X	X	X	X	X	X	X	X
HABITATION MODULE 1			X	X	X	X	X	X	X
HABITATION MODULE 2				X	X	X	X	X	X
LOGISTICS MODULE					X	X	X	X	X
TUNNEL						X	X	X	X
STAGE ASSEMBLY							X	X	X
RMS/CONTROL								X	X
CONSTRUCTION FACILITY									X

FIGURE 3.23 SOC BUILDUP SEQUENCE

Table 3.14 lists the mass properties for selected SOC buildup configurations. Cases 1, 4, 29 and 30 were used to determine the spread of momentum control requirements between partial and full up SOC configurations.

3.4.4 Control Modes Selection

Both the method and the degree to which the SOC is controlled in attitude depends on the buildup scenario -- direct, or evolutionary. In either case the control requirements for SOC during orbiter docking are the same; however, the method of implementing the requirements depends on the system hardware used in both docking and free flight operation.

Control of the SOC attitude appears to be feasible by use of large CMGs. Two existing double-gimballed CMG designs are the Bendix Skylab model, at 2300 ft-lb-sec (3100 N-M-sec), and the Sperry Model 4500, at 4500 ft-lb-sec (6100 N-M-sec). Both of these were considered valid candidates early in the study; however, when the momentum magnitudes that could potentially be encountered by SOC were determined, a somewhat larger component appeared to be needed. Since the Sperry design is a later technology than that of the Skylab, the new design was postulated as an improved Sperry design that could deliver up to 10,000 ft-lb-sec (13,500 N-M-sec) with a weight of 800 lb (360 Kg), up from 650 lb (293 Kg), and an average electrical power of 125 watts (no change from previous design). By using advanced materials technology and an optimized rotor shape the rotor speed may be increased, thus the size of the improved CMG is essentially the same as the present Model 4500.

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TABLE 3.14 MASS PROPERTIES FOR SELECTED CONFIGURATIONS

MASS PARAMETER	CASE			
	4	1	29	30
WEIGHT (LB)	42,174	217,898	223,619	399,337
MASS (SLUG)	1310.8	6772.5	6950.3	12,411.8
CENTER OF GRAVITY (FT)				
X	23.60	-2.15	38.28	17.77
Y	-18.79	-3.64	0.20	0.11
Z	1.30	-31.67	4.77	-14.75
MOMENTS OF INERTIA ($SLUG-FT^2$)				
Ixx	2,358,918	10,875,070	9,849,321	22,240,950
Iyy	253,916	9,210,563	5,491,327	24,261,450
Izz	2,498,368	4,754,129	7,445,907	14,894,630
Ixy	-292,189	-340,095	1,609,025	1,638,761
Ixz	15,398	1,606,598	-105,600	6,457,639
Iyz	-60,726	-757,388	-186,395	-166,005
PRINCIPAL MOMENTS OF INERTIA ($SLUG-FT^2$)				
Ixx	2,393,551	11,394,370	10,387,570	26,733,720
Iyy	212,634	9,174,568	4,953,299	23,596,310
Izz	2,505,016	4,270,826	7,445,684	11,067,000
PRINCIPAL AXIS OFFSET ANGLES (DEGREES)				
θ_x	-1.491	4.193	-3.318	19.381
θ_y	-12.434	14.701	-3.090	24.452
θ_z	7.421	14.010	-18.304	-32.080

CASES: 4 = SERVICE MODULE ALONE
1 = SERVICE MODULE WITH ORBITER DOCKING IN "A" ORIENTATION
29 = COMPLETED SOC ALONE
30 = COMPLETED SOC WITH ORBITER DOCKED IN "A" ORIENTATION

NOTE: PRINCIPAL AXIS OFFSET ANGLES COMPUTED USING XYZ ROTATION SEQUENCE

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The SOC orientations discussed in this section are best described in terms of the orbit plane. An axis of the SOC which is perpendicular to the orbit plane is referred to as "POP" such as X-POP or Y-POP, etc. The orientation may be further limited by indicating an axis which is vertically oriented if the SOC is flying in an earth-centered attitude (local vertical). Thus, Y-POP, ZLV indicates that the SOC is flying with its Y axis perpendicular to the orbit plane, its Z axis pointed vertically, and its X axis along the velocity vector throughout an orbit. An inertially fixed flight mode may be referred to as Y-POP-inertial, indicating that the X and Z axes remain fixed in space during each orbit revolution.

The analytic approach used in determining the SOC control mode is shown in Figure 3.24. In the direct buildup sequence, the first service module delivered to orbit merely waits for the orbiter to return, and can be oriented to use minimum control. Since no payloads are aboard, the system may be powered down and the solar array gimbals locked if desired. A predetermined balance between gravity gradient and aerodynamic disturbance torques may be maintained during this quiescent period using the capability of only a small momentum control system. The orbiter may either dock to the spacecraft while it is in this skewed orientation or the SOC may temporarily reorient with its X-axis along the velocity vector using its CMGs or the RCS. This is indicated in the upper left of the figure.

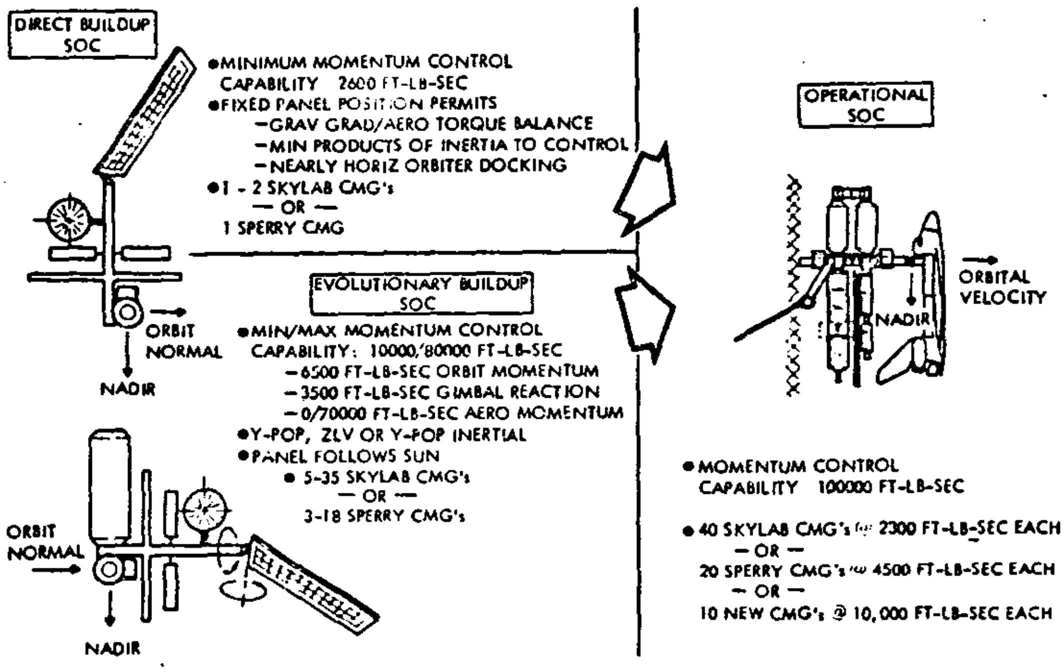

FIGURE 3.24 ATTITUDE CONTROL SYSTEM BUILDUP - EVOLUTIONARY VS DIRECT

In the evolutionary buildup sequence the first service module will be capable of supporting payloads, and thus must orient to provide preplanned solar array motion. Y-POP, ZLV, or Y-POP inertial are the simplest attitudes to analyze, but small offsets from this orientation to reduce disturbances are also apporpriate. Large offsets may shade the solar panel.

It is therefore clear that greater control capacity could be required for the evolutionary buildup case than for direct buildup. The control capacity for the evolutionary buildup case has been estimated conservatively in the lower left of the figure. Gravity gradient momentum is estimated to be approximately in the range of the orbit momentum. Solar array motion generates gimbal reactions from gyroscopic and friction torques, which contributes a second component of disturbance momentum. Aerodynamic drag on the single solar array develops a third and major component of disturbance momentum. The resulting CMG requirement may approach that of the completed SOC, which is described on the right of the figure, but small orientation biases toward the principal axes and in a direction to counter solar array drag will reduce the requirement to about half of that required for the completed SOC. Thus, each service module may contain half of the momentum control capability required for a complete SOC, and be capable of meeting nearly all of the foreseeable control requirements for the various SOC buildup options and interim mission operations.

3.4.5 CMG Sizing

A more precise analysis using digital simulation techniques was performed to determine disturbance effects on the four selected cases. To ensure that the maximum conditions were examined both inertial and local attitude holds were simulated.

The computer program used in the analysis is a general purpose program written in the Continuous System Modeling Program (CSMP III) language. It includes detailed models of gravity gradient, aerodynamic, solar pressure and earth magnetic field disturbances; however, only the first two disturbance models were used in the SOC analysis. Mass and aerodynamic properties for the SOC body, solar arrays, and orbiter (where appropriate) were entered separately. The solar arrays are rotated to constantly face the sun and their instantaneous position is used to generate the total mass properties and aerodynamic areas for that instant. The sun was assumed to lie in the orbit plane for all runs. Aero torques would be reduced for other sun angle geometries. The local vertical cases were oriented with the body axes aligned to the orbit axes (Y-POP, ZLV), which allows gravity gradient torques to be generated solely from the products of inertia. The inertial attitude hold cases start in the same orientation, but the X and Z body axes rotate relative to the orbit axes which allows both the the basic moments and the products of inertia to generate gravity gradient torques.

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A summary of the results of the computer runs are shown in Figures 3.25 through 3.32. In each figure the torques and momentum are shown separately for aerodynamic and gravity gradient disturbances in the first two columns, then at the right most column the combined effects on torque and momentum are shown. Momentum control sizing and RCS momentum dumping frequency is determined from the combined effects.

Table 3.15 describes the cyclic and secular momentum components. In general, the addition of the orbiter tends to increase both cyclic and secular disturbances. However, the effect of the orbiter on the products of inertia and aerodynamic moment arm tend to confuse the trend. In most cases the previously estimated CMG sizing is easily capable of momentum control for prolonged periods without becoming saturated. All cases could benefit from the use of attitude biases which reduce secular momentum, especially configuration 30 in local vertical flight. It may also be appropriate to use SOC on board software to predict optimum attitude biases for each change in mass and aerodynamic properties.

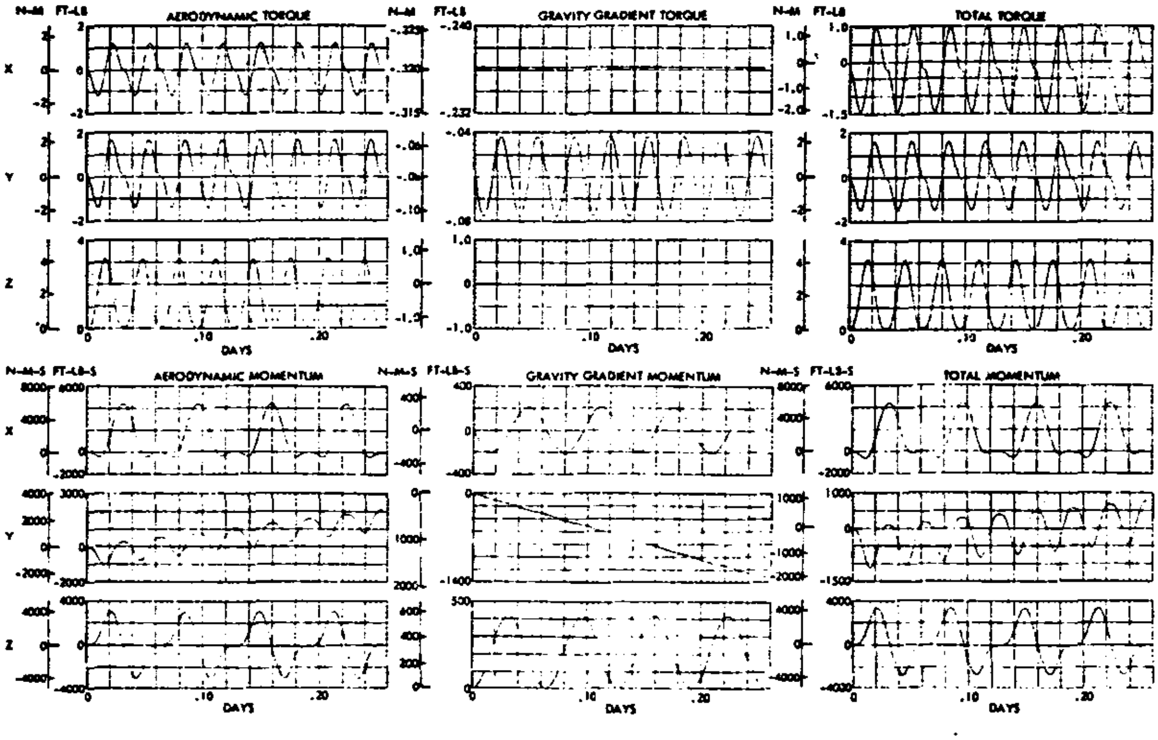

FIGURE 3.25 CONFIGURATION 4 - MOMENTUM CHARACTERISTICS IN LOCAL VERTICAL ORIENTATION

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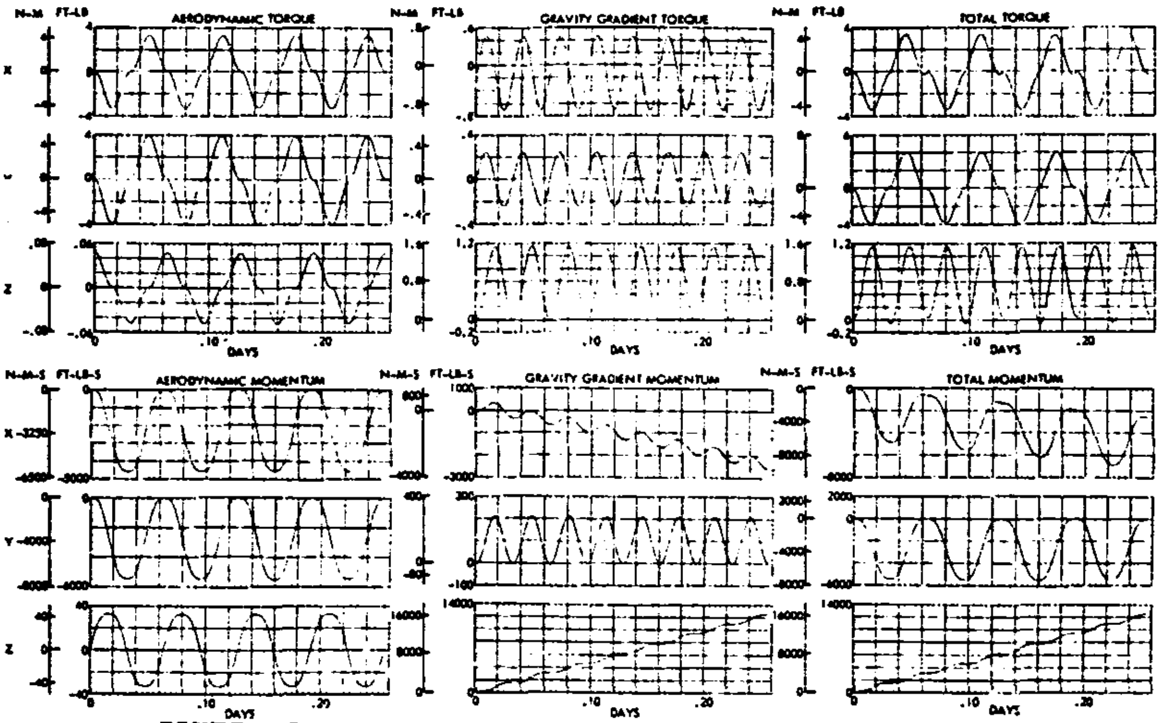

FIGURE 3.26 CONFIGURATION 4 - MOMENTUM CHARACTERISTICS IN INERTIAL ORIENTATION

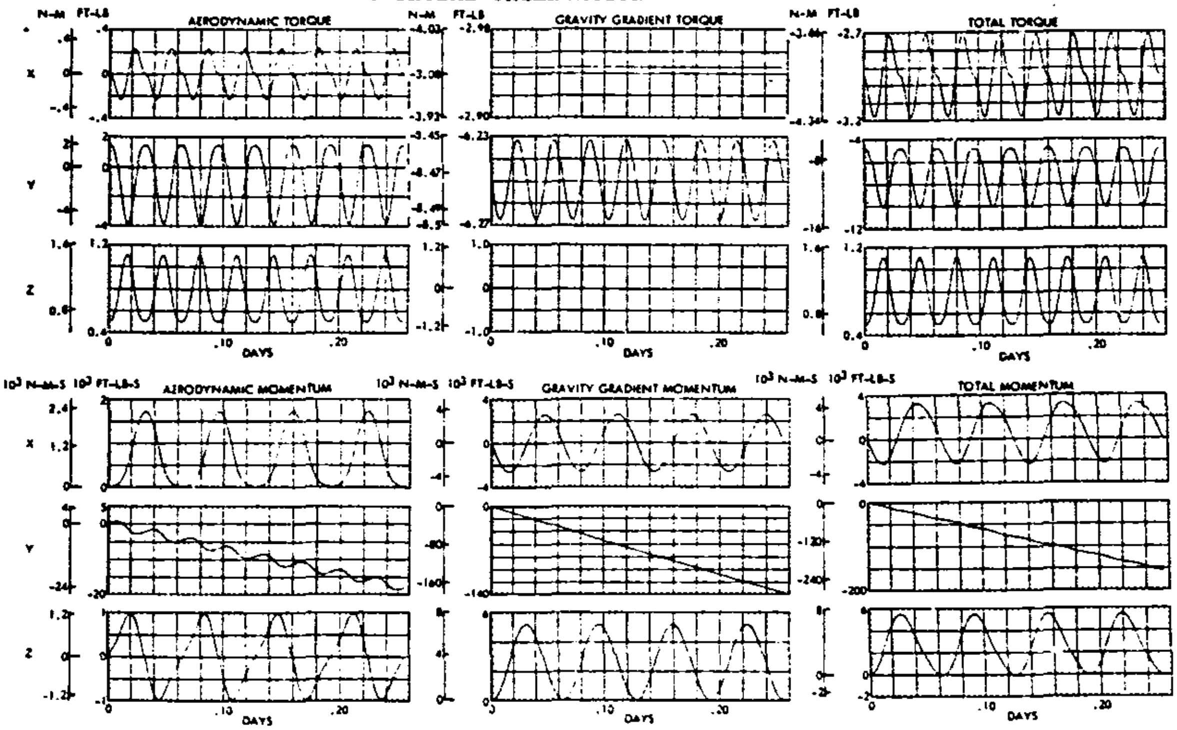

FIGURE 3.27 CONFIGURATION 1 - MOMENTUM CHARACTERISTICS IN LOCAL VERTICAL ORIENTATION

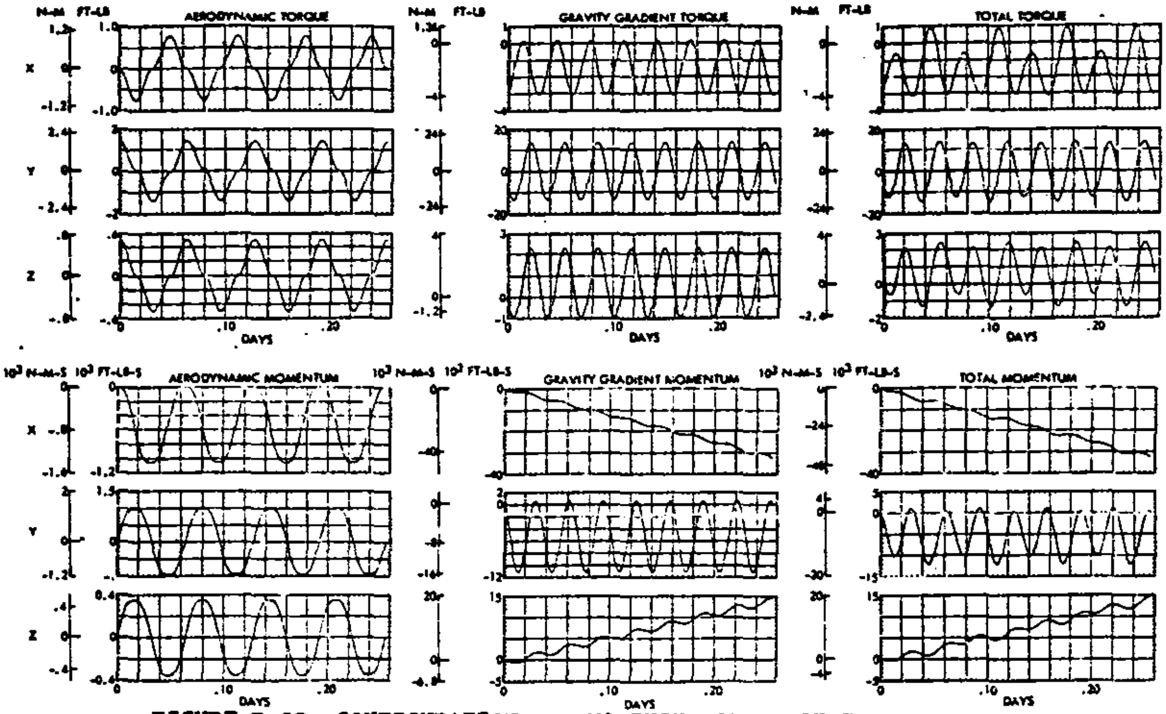

FIGURE 3.28 CONFIGURATION 1 - MOMENTUM CHARACTERISTICS IN INERTIAL ORIENTATION

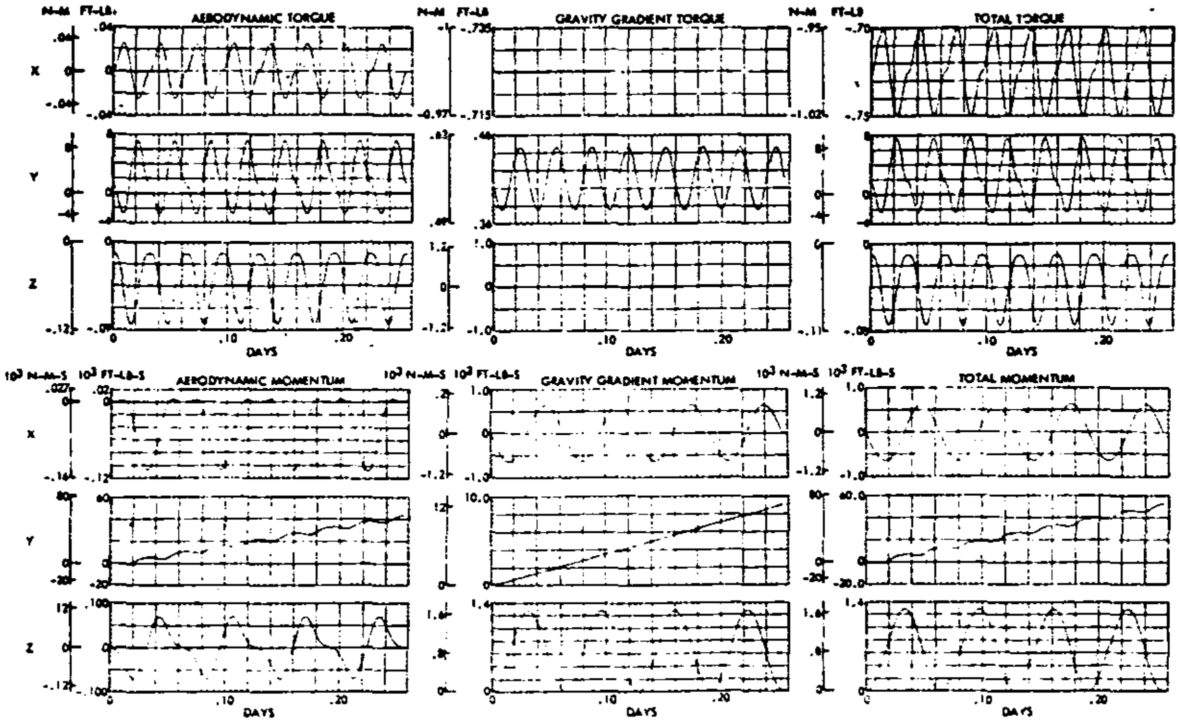

FIGURE 3.29 CONFIGURATION 29 - MOMENTUM CHARACTERISTICS IN LOCAL VERTICAL ORIENTATION

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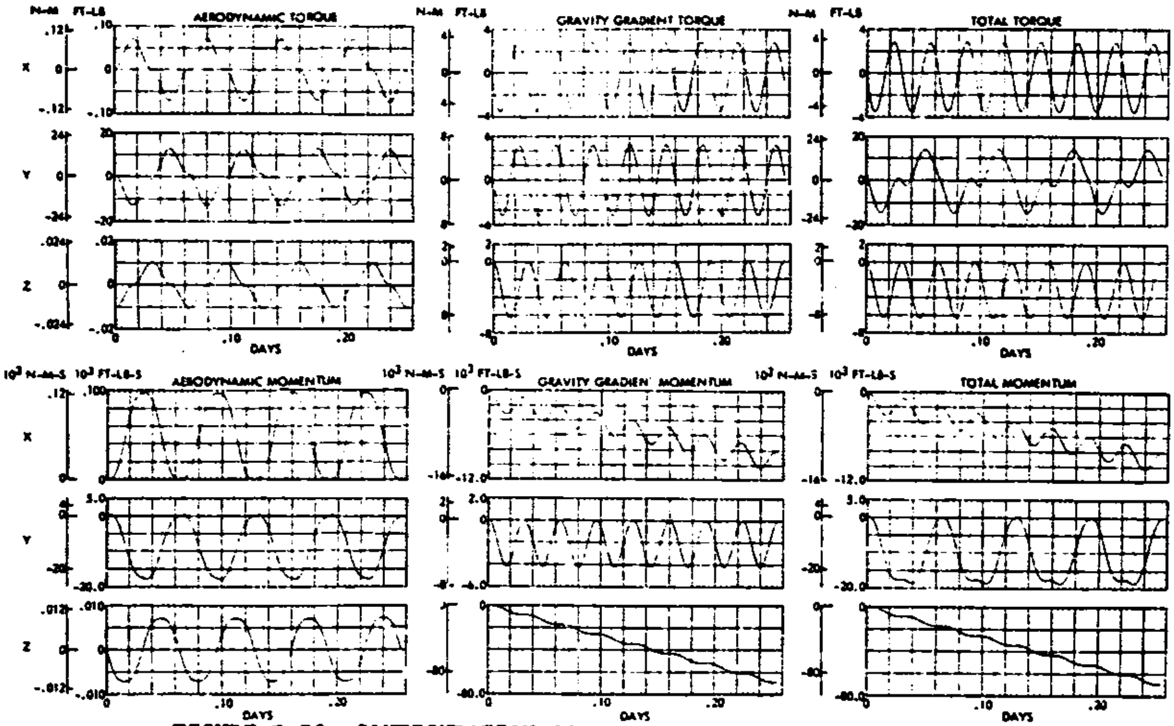

FIGURE 3.30 CONFIGURATION 29 - MOMENTUM CHARACTERISTICS IN INERTIAL ORIENTATION

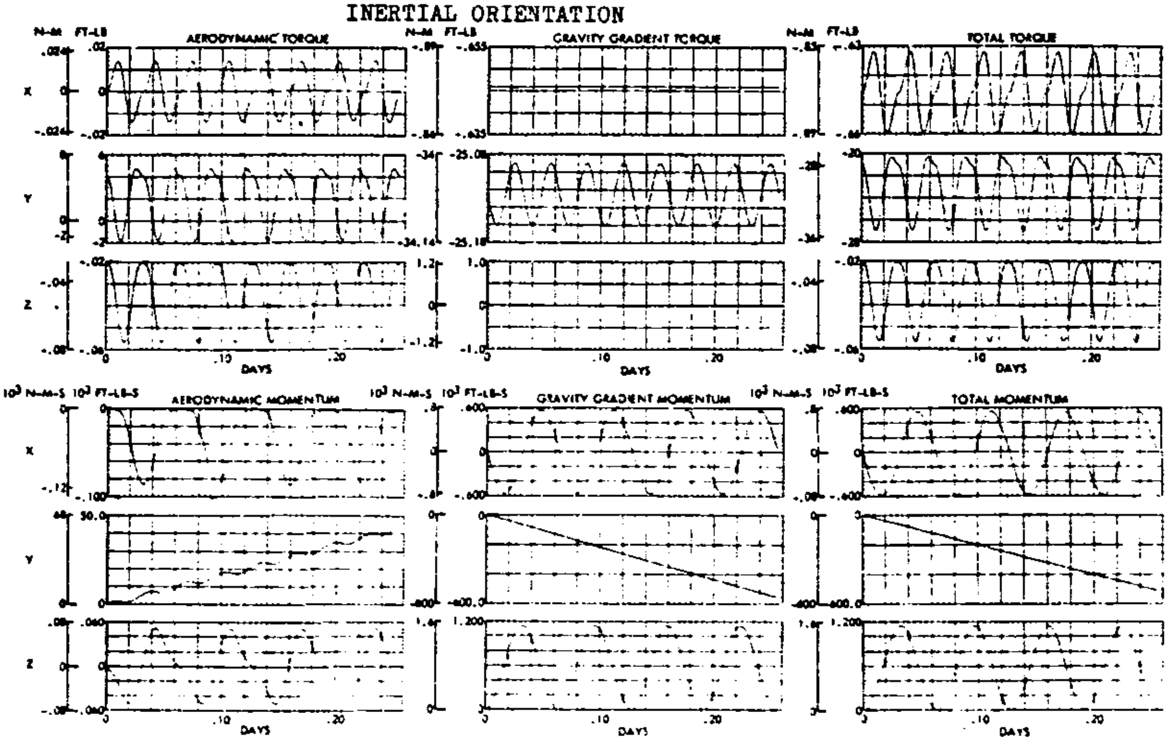

FIGURE 3.31 CONFIGURATION 30 - MOMENTUM CHARACTERISTICS IN LOCAL VERTICAL ORIENTATION

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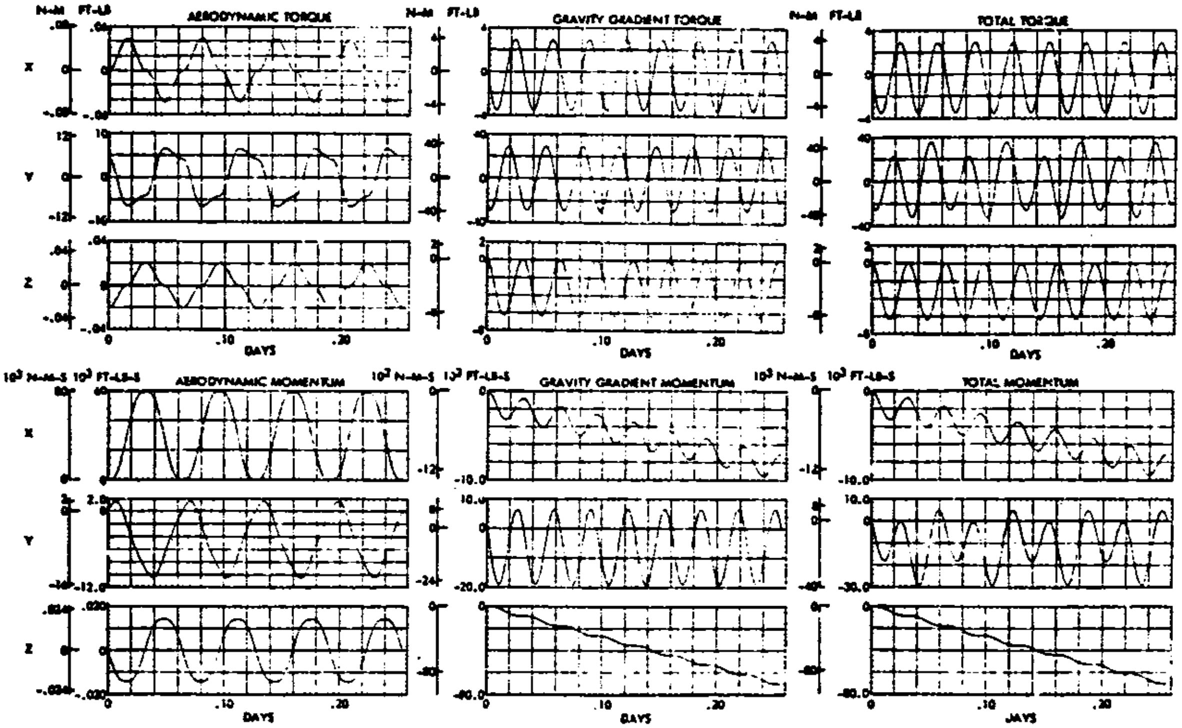


FIGURE 3.32 CONFIGURATION 30 - MOMENTUM CHARACTERISTICS IN INERTIAL ORIENTATION

TABLE 3.15 SUMMARY OF DISTURBANCE MOMENTUM COMPONENTS

	ORIENTATION →	LOCAL VERTICAL		INERTIAL	
CONFIG. ↓	MOMENTUM → COMPONENT	CYCLIC (FT-LB-SEC)	SECULAR (FT-LB-S/ORB)	CYCLIC (FT-LB-SEC)	SECULAR (FT-LB-S/ORB)
4 (SERVICE MODULE ONLY)		4000	200	3,500	3,000
1 (SERVICE MODULE AND ORBITER)		6000	39,000	6,000	9,500
25 (COMPLETED SOC ALONE)		500	12,000	9,000	18,000
30 (COMPLETED SOC AND ORBITER)		600	140,000	15,000	18,000

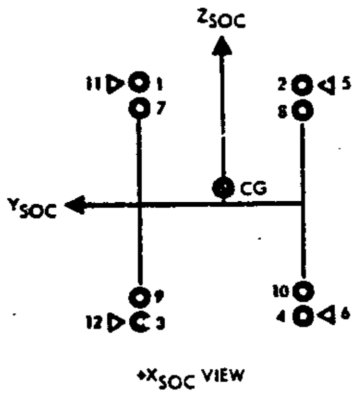
3.4.6 RCS Thruster Arrangement

The baseline RCS thruster arrangement for the completed SOC configuration (Reference 4) is pictured schematically in Figure 3.33. The geometry and number of thrusters provide orbit makeup propulsion either independently or in conjunction with pitch or yaw momentum dumping. A propellant feed arrangement (string A or string B) is also identified in the figure to enable full control even when one of the two supply systems or a thruster has failed. The configuration utilizes two RCS boom assemblies, one on each service module to meet the requirements.

Until the second service module is mated and operational in the direct buildup case, the RCS on the first service module must be capable of at least periodically dumping the momentum from the CMGs. This requires three-axis torquing with full thruster and propellant feed system redundancy, preferably without propellant-consuming interaxis coupling problems. Orbit makeup maneuvers are probably not needed. It should be possible to assemble the SOC at orbit altitudes sufficient to meet reasonable decay life requirements without orbit makeup. However, orbit makeup capability would certainly enhance operational margins and reduce program risk factors.

To meet the basic requirements for direct buildup some modifications to the baseline system are required. With just half the system on each service module there is no redundancy in roll control for single service module operations during buildup. Also, for Shuttle tended operations during subsequent revisits the combined c.g. is dangerously near the -Z displacement limit which could cause the loss of pitch-up control authority. Thus, some form of added RCS capability is needed. Perhaps the simplest arrangement meeting the basic functional needs is the addition of thrusters to the baseline RCS clusters. This approach is illustrated in the schematic of Figure 3.34. Ten additional thrusters have been added to the six in the original baseline configuration. Moment couples are provided in roll and pitch, but yaw torques are accomplished using one-sided thrusting. One-sided yaw thrusting also produces a pitching moment when the c.g. is offset along the vertical axis. Large offsets, such as during Shuttle tending operations, can lead to severe fuel inefficiencies because significant firing of the pitch thrusters is required to compensate for the yaw/pitch coupling.

There are other solutions to the RCS problem during SOC buildup, such as the installation of a small auxiliary boom and appropriate thruster sets to provide three-axis control torques (and orbit makeup, if desired). In their most austere form these RCS arrangements may be inadequate for evolutionary build plans where a single service module is required to perform in more than just a between-Shuttle-visit survival mode. Here, it may also be required to provide both orbit makeup and three-axis momentum dumping in support of various early mission operations.



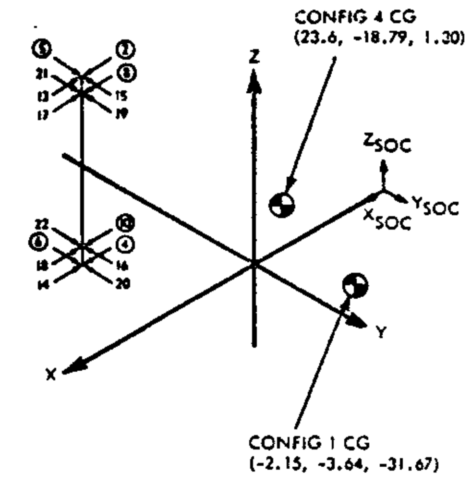
STRING A	1, 2, 3, 4, 5, 6
STRING B	7, 8, 9, 10, 11, 12

	ATTITUDE CONTROL		
+PITCH	2 & 1	OR	7 & 8
-PITCH	3 & 4	OR	9 & 10
+YAW	2 & 4	OR	8 & 10
-YAW	1 & 3	OR	7 & 9
+ROLL	6 & 11	OR	6 OR 11
-ROLL	5 & 12	OR	5 OR 12

NOTE: CENTER OF MASS MUST BE << 40 FT FROM ORIGIN OF CONTROL AXES SINCE COUPLES ONLY IN ROLL WITH NO FAILURES.

DELTA VEE MANEUVERS (+X)
1, 2, 3, 4 OR 7, 8, 9, 10

FIGURE 3.33 BASELINE SOC RCS THRUSTER ARRANGEMENT



- THRUSTER AND SYSTEM REDUNDANCY
- ORBIT MAKEUP NOT REQUIRED
- TEMPORARY USE

AXIS	PRIMARY	BACKUP
+ ROLL	6, 15	19, 22
- ROLL	5, 20	16, 21
+ PITCH	2, 14	8, 18
- PITCH	4, 13	10, 17
+ YAW	2, 4	8, 10
- YAW	13, 14	17, 18

16 THRUSTERS NEEDED FOR ADEQUATE SINGLE SERVICE MODULE CONTROL IN DIRECT BUILDUP SEQUENCE

- RCS LOCATED AT (35, -30, ±35)
- ○ EXISTING BASELINE THRUSTERS

FIGURE 3.34 SMALL RCS FOR DIRECT BUILDUP SOC

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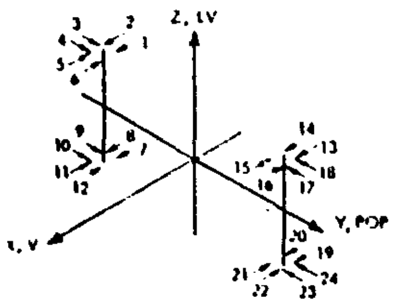
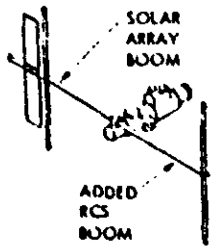
The main point is that the baseline RCS configuration is inadequate, even for the direct buildup case, and some modifications are required. There are minimum mods which can satisfy the direct buildup, but which may not meet the needs of many evolutionary build plans. To meet the full early mission needs with a single service module the auxilliary boom mentioned above could be enlarged to the extent necessary for adequate control as shown schematically in Figure 3.35. The auxilliary boom would be used until the second service module is mated and its RCS is operational. The thruster arrangement shown in the figure provides +X translational thrusting without angular disturbances if the thrusters are pulsed in a coordinated manner to account for c.g. offsets. This also applies to pitch and yaw momentum dumping. When the orbiter is present only momentum dumping functions need be undertaken and moment couples in all axes may be used to minimize plume impingement on the orbiter. The primary and backup thruster groups pictured in the figure may utilize completely separate propellant feed systems. The main disadvantage of this arrangement is the added complication of the temporary boom.

To avoid this complication the possibility exists for the service module design to be changed to allow the final RCS configuration (complete baseline system) to be installed on the first service module. Thus, the full-up RCS capability would be available from the beginning and would satisfy all of the evolutionary buildup options. This would also save the deadended development of the temporary RCS boom configuration. Further, it would eliminate a propellant fill connection through the mating interface between the two service modules and it would facilitate dual propellant feed systems between RCS clusters. Although this approach has obvious packaging problems for fitting into the orbiter bay, the temporary boom approach would also face this constraint. Further study is warranted.





- NO INTERAXIS COUPLING
  - INDIVIDUAL THRUSTER MODULATION REQ'D
- +X ORBIT MAKEUP THRUSTING
- PITCH AND YAW CONTROL CONTRIBUTES TO ORBIT MAKEUP FOR MOST CONFIGS WITH ORBITER ABSENT
- PITCH, YAW, ROLL MOMENT COUPLES FOR ADVERSE CG AND ORBITER PRESENT

- CONCLUSION
EFFICIENT & ADEQUATE DESIGN

	ORBIT MAKEUP + MOMENTUM DUMP		MOMENTUM DUMP	
AXIS	PRIMARY	BACKUP	PRIMARY	BACKUP
+ROLL	9, 18	10, 17	9, 18	10, 17
-ROLL	3, 24	4, 23	3, 24	4, 23
+PITCH	1, 14	2, 13	1, 12, 14, 21	2, 11, 13, 22
-PITCH	7, 20	8, 19	6, 7, 15, 20	5, 8, 16, 19
+YAW	1, 7	2, 8	1, 7, 15, 21	2, 8, 16, 22
-YAW	14, 20	13, 19	6, 12, 14, 20	5, 11, 13, 19
+X	4, 20	2, 8, 13, 19	1, 7, 14, 20	2, 8, 13, 19

FIGURE 3.35 RCS COMPONENTS -- ADD-A-BOOM

References

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4.0 SOC RESUPPLY & FUEL TRANSFER

This task consists of two subjects as indicated by the heading. The SOC resupply task was concerned with the resupply of logistics to support the normal operations of the SOC and the logistics required for construction and flight support operations. The fuel transfer task concentrated on the transfer of propellant for servicing an OTV.

4.1 SUMMARY

The objective of this task was to determine the requirements imposed on the SOC and on the orbiter to support the SOC resupply logistics operations for the normal housekeeping and operational activities, and to determine the impacts associated with the transfering of propellant from the orbiter to SOC storage or directly to an OTV.

The SOC resupply issues were concerned with the development of the SOC logistics module (LM) exchange procedure, and the capability of the orbiter to transport a full SOC crew of eight.

The fuel transfer analysis evaluated the fuel transport and transfer concept developed by the General Dynamics Corporation as it would apply to this SOC operation. Suggested improvements/revisions were defined as appropriate for the SOC operation.

<u>No Special Equipment Required to Exchange SOC LM</u>

The development of a logistics module exchange procedure was necessary because a single attach port only was dedicated to accommodate the logistics module (LM). Consequently, the attach port must be vacated before a full logistics module is installed. This requirement necessitates the incorporation of a parking/holding position for either the empty or the full LM during the exchange operation. The analysis indicated that the utilization of the handling and positioning aid (HPA) located on the right side of the orbiter on the forward section of the payload bay longeron very adequately accomplishes this parking/holding requirement. The HPA has been identified as a necessary device for SOC assembly and for holding construction projects during assembly (Reference 6). Consequently, at this time the HPA can be considered as a piece of equipment that is part of the standard available equipment for space operations.

The orbiter RMS has the reach and motion capability necessary to perform the LM exchange procedure. The exchange procedure, therefore, consists of the extraction of the LM, by the RMS, from the orbiter payload bay and placing it on the HPA in a holding position. The RMS then removes the spent logistics module from the SOC and returns it to the orbiter payload bay. The full LM is removed from the HPA, transported, and berthed to the vacated, dedicated, LM port. The utilities interfaces are remotely actuated as described in Section 2.0 to complete the LM exchange procedure.

No Special Equipment Required to Inatall or Exchange Construction and Flight Support Logistics Cradles

Similar module exchange conditions exist for handling of logistics cradles containing space construction materials. The exchange procedure is identical to the SOC LM exchange procedure previously described. The HPA is again used as a parking/holding device for the cradle. However, because of the long reach necessary to deposit a construction cradle at the construction site, the SOC remote control manipulator must perform the transportation phase of the exchange procedure. Consequently, no additional/special equipment is required to support this logistics cradle exchange.

The Orbiter Can Transport a Full Crew of Eight

Various scenarios can be generated that require the transportation of eight crew persons to and from the SOC. The orbiter as the transportation vehicle must provide this capability. Two orbiter arrangements were considered to verify the orbiter capability. The standard cabin seating arrangement has accommodations for 10 people in a rescue mode. If the operation of the orbiter is assumed to be accomplished by the commander and pilot only, the eight remaining seats can be occupied by the SOC crew. However, if the mission on which the crew is being transport requires three or four orbiter crew, then one or two additional seats must be provided in the mid deck section of the cabin. These additional seating positions can be accommodated by removing appropriate portions of the forward modular locker stowage compartments with the airlock inside the cabin. If the airlock is moved outside the cabin or the docking module is acting as an airlock, the modular storage lockers can remain.

Adequate storage for personal belongings and for emergency provisions appear adequate for all arrangements investigated. Clear paths for emergency exit for a pad abort condition are also available. Additional analysis and tests are necessary to positively verify the emergency egress procedures. However, this preliminary analysis indicates the feasibility of the orbiter to transport a full crew of eight to or from the SOC.

Low-Risk Technology Is Available For Zero-G Fluid Transfer

A conservative system design, based partly on concepts recommended by General Dynamics in Reference 1, was developed for zero-g transfer of LOX and LH_2 from the orbiter to OTVs berthed on SOC (Figure 4.1). The features of this system include a dewar-type supply tank in the orbiter, helium pressurized transfer, capillary screen-channel propellant acquisition, and pressure/vent cycling for prechilling of the LH_2 receiver tank. Overall propellant losses, from ground loading to OTV propulsive usage, were estimated to be approximately 7.5%.

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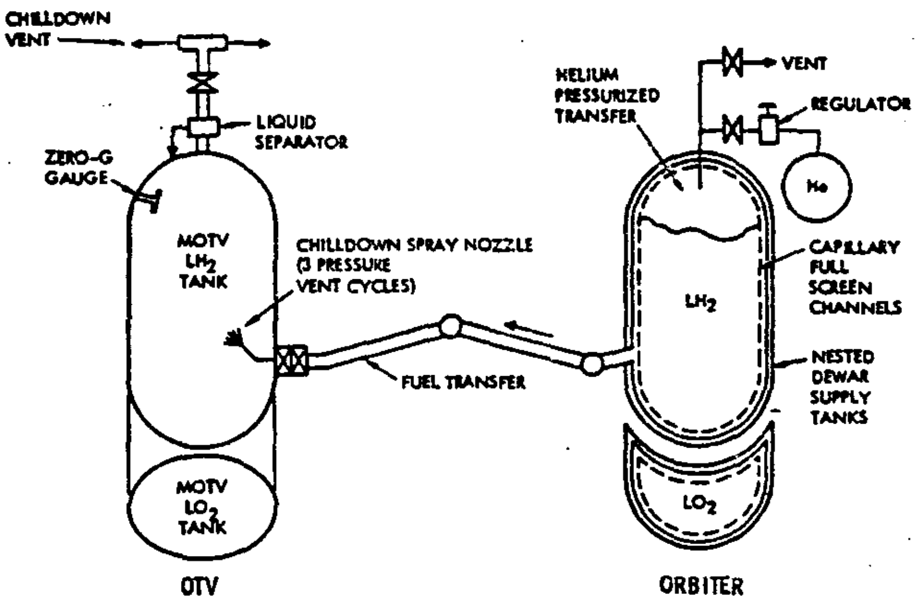


FIGURE 4.1 SOC-GDC BASELINE REFUEL SCHEMATIC

Advanced Concepts Can Substantially Reduce Transfer Losses

An advanced transfer system concept was also developed which has the potential of reducing overall losses to less than 2.5%. This includes a number of candidate refinements such as a lightweight supply tank with multi-layer insulation (MLI) instead of the heavier dewar, subcooling of propellants to reduce boiloff losses, pumped transfer for reduced tank pressure requirements, autogenous pressurization to reduce pressurization system weight, non-venting pre-chill of the receiver tanks to eliminate chilldown losses, simplified capillary devices to reduce tank cost and weight, and centralized SOC transfer control.

The technique recommended for zero-g propellant guaging is acoustic resonance, with ullage compliance and RF guaging retained as fall back approaches.

Remote Controlled Fluid Transfer Is Viable

For fluids which require resupply on a regular basis, e.g., LOX, LH_2 and hydrazine, transfer by means of a remotely operated disconnect through a single filling point into one or more stowage tanks on SOC is the recommended mode of resupply. This approach minimizes EVA requirements and the hazards associated with exposure to spillage and leakage. For fluids requiring infrequent resupply, such as fluorocarbons, container replacement using the RCM and manually operated disconnect fittings is considered a practical approach.

Recovery of ET Propellant Residuals and Storage on SOC Shows Major Benefits

An analysis was made of cryogenic propellant storage on SOC, in combination with sub-orbital recovery of residuals from the Shuttle External tank (ET), and propellant payload sharing on under-loaded orbiter flights. The results indicated that a drastic reduction (1/3 to 1/2) can be achieved in the total number of Shuttle launches required for a typical yearly SOC traffic schedule.

OTV Propellant Storage on SOC Is Recommended

First, the ability to store OTV propellants on the SOC can save on the number of propellant logistics flights required to support OTV operations. These savings result from (1) the recovery of ET unused propellant, (2) the elimination of "round-off" flights - propellant needed for a given OTV mission above an integer number of Shuttle flights, and (3) basic reductions in overall OTV propellant requirements which can be achieved through lightweight space based OTV designs. Further savings would be possible through reduced propellant losses by incorporating active refrigeration into the propellant storage facility.

Second, propellant storage on SOC could also uncouple Shuttle logistics from SOC based OTV operations. Instead of requirements for breaking into the Shuttle manifesting plan for a cluster of three, four, or possibly five closely spaced flights in support of an MOTV mission, the propellant could be delivered to SOC on a routine scheduled basis, thus, easing fleet management and potentially improving fleet utilization. Propellant storage could also provide a rapid response capability for rescue and/or other high value services.

Because of these major benefits, OTV propellant storage on SOC is recommended.

Active Refrigeration Is Feasible

Analysis shows that active refrigeration of the above LOX and LH_2 storage tanks on SOC is practical and beneficial in terms of reducing storage losses and providing subcooled propellant. The latter can improve OTV structural mass fraction and reduce mission boiloff losses. It is particularly attractive in conjunction with the recovery of residual propellants from the Shuttle ET.

4.2 SOC LOGISTICS MODULE AND LOGISTICS CRADLES EXCHANGE PROCEDURE

An analysis of this operation for both the LM and the logistics cradles was performed in order to determine the equipment required, the orbiter and SOC impacts, and the operational sequence(s).

Figure 4.2 illustrates the SOC configuration in a full-up operational arrangement, and indicates the location of the SOC LM and of the logistics cradles supporting space construction and flight support facility operations.

SOC Logistics Module

Only one port on the reference SOC configuration is dedicated to accept the SOC logistics module. Consequently, when exchanging the spent module with a full module, a temporary holding/parking position is necessary. This holding position can either be on the SOC or on the orbiter. When the SOC is in full operation all ports are dedicated to support the SOC operations. Consequently, the orbiter must provide this capability.

The issues involved in determining the exchange procedure(s) are listed in Table 4.1. Three functions are identified that affect this analysis (1) deployment of the LM from the payload bay, (2) transportation of the LM to the SOC, and (3) holding/parking the LM. Implementation of these functions determines the equipment required. Table 4.1 indicates the equipment options that are available to perform these functions. The payload installation and deployment aid (PIDA) was selected to perform the deployment of the LM from the payload bay. This concept was selected over a rotating flight support cradle type device (MMS/FSS) because it not only was identified in Section 3.0, SOC assembly, as the selected method for deploying the SOC modules from the payload bay for the SOC assembly operation but was also the least complex mechanism and had the least impact to the shuttle. The orbiter's RMS has the reach capability to transport the LM from the orbiter to the SOC. The principal issue was the determination of a holding/parking position and the equipment required for this function. Three exchange procedure options were developed and analyzed in order to determine the equipment required. Figure 4.3 illustrates the options and Figure 4.4 schematically represents the exchange sequences for each option.

Each option is identified by the number of transport equipment required and the type of holding/parking device required. The first option utilizes the orbiter RMS to perform the transport function with the handling and positioning aid (HPA) providing the LM parking function. The exchange sequence, indicated in Figure 4.4, has the full LM removed from the PIDA deployed position by the RMS and transported to the HPA holding/parking position. The RMS continues to the SOC, engages the spend LM and deposits the LM on the PIDA device which then stores the module. The RMS retrieves the full LM from the HPA and berths it to the SOC, thus, completing the exchange.

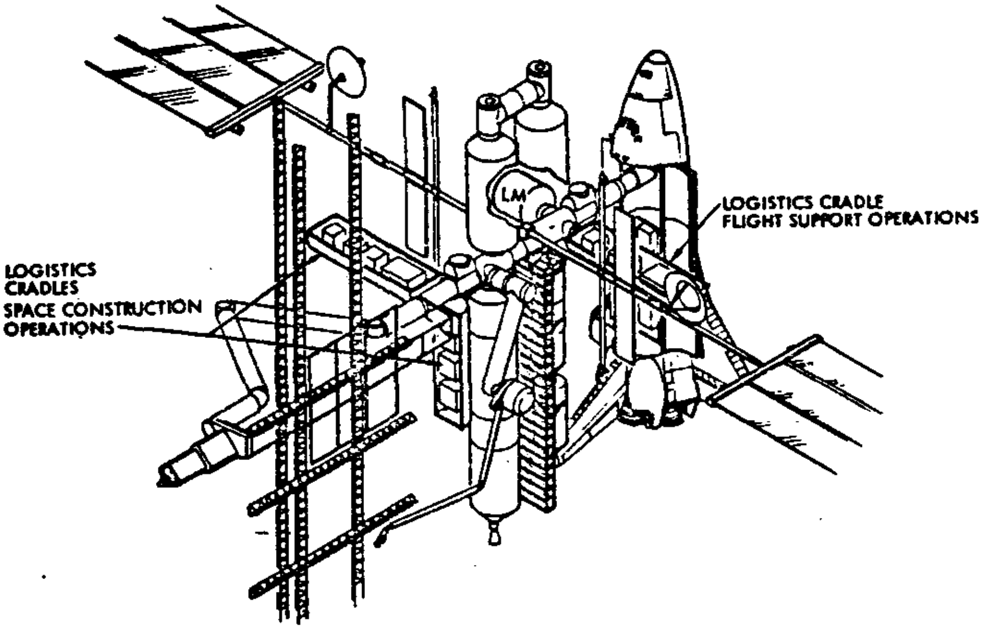


FIGURE 4.2 SOC CONFIGURATION

TABLE 4.1 LOGISTICS MODULE/CRADLE CHANGEOUT ISSUES

• EQUIPMENT REQUIRED	- DEPLOY FROM PAYLOAD BAY	PIDA MMS/FSS
	- TRANSPORT TO/FROM SOC	RMS RCM
	- PARKING	HAPA MMS/FSS RCM
• REACH	- RMS	
	- RCM	
• ORBITER ORIENTATION		

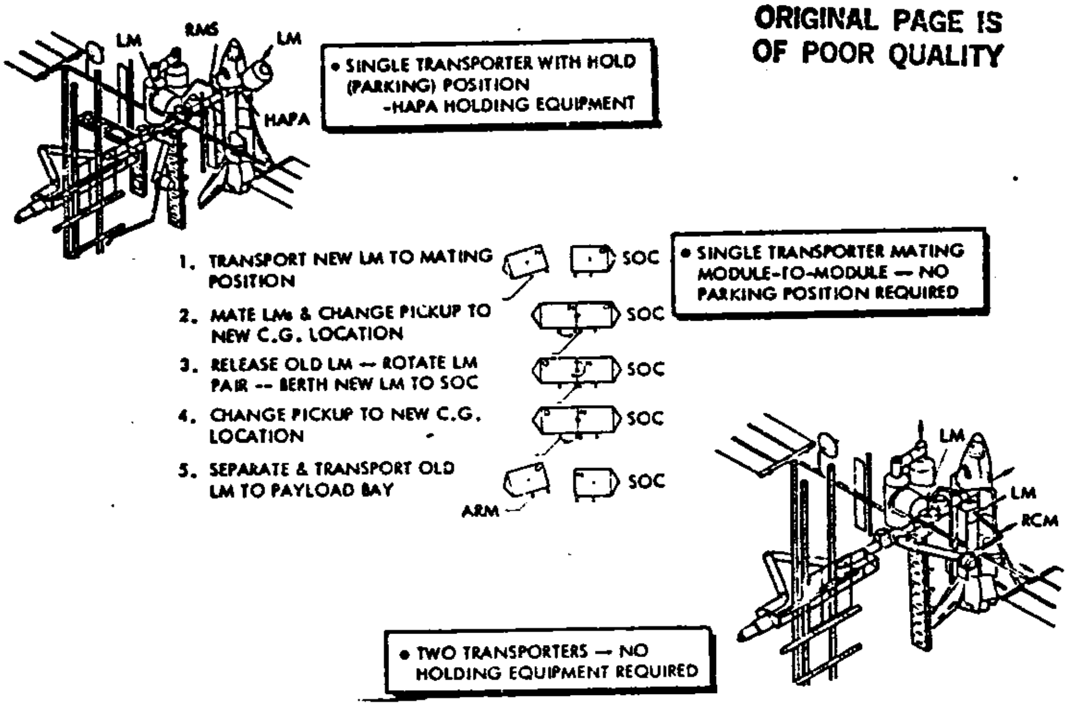
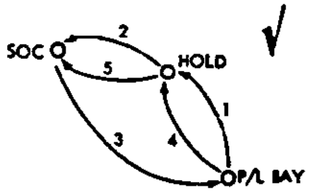
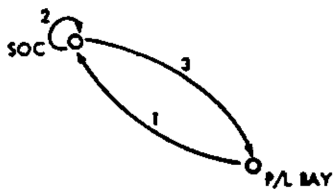


FIGURE 4.3 SOC LOGISTICS MODULE EXCHANGE OPTIONS

- SINGLE TRANSPORTER
  1. NEW LM TO HOLD
  2. MOVE TO SOC (TRANSPORTER)
  3. OLD LM TO PAYLOAD BAY
  4. MOVE TO HOLD (TRANSPORTER)
  5. NEW LM TO SOC



- MODULE TO MODULE MATING
  1. NEW LM TO SOC MODULE
  2. EXCHANGE LM BERTHING
  3. OLD LM TO PAYLOAD BAY (NO EXCESS TRANSPORTER MOVEMENT)



- TWO TRANSPORTERS
  1. RMS MOVE NEW LM TO HOLD
  2. RCM MOVE OLD LM TO PAYLOAD BAY
  3. RMS MOVE NEW LM TO SOC

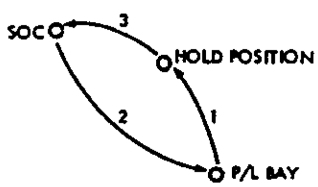


FIGURE 4.4 EXCHANGE SEQUENCES

The second option, shown in Figure 4.3, utilizes the RMS to transport the full LM from the PIDA deployed position and berthing the full LM to the spent LM. The spent LM is released from the SOC port and the assembly, the spent LM and full LM, is rotated to allow the full LM to berth to the SOC port. The spent LM is separated from the full module and returned to the orbiter payload bay. The RMS performs all of the transport and rotation functions. This concept requires an additional berthing device on the LM in order to accomplish the mating of the spent and full LM. Additional command signals will be required across the LM port in order to activate the mating port between the two LMs. No holding/parking device is required for this option, however.

The third option utilizes both the RMS and the SOC remote control manipulator (RCM). The RMS performs the function of transporting the full LM from the PIDA deployed position to the SOC. However, it holds the LM while the RCM removes the spent LM and transports it to the payload bay. This procedure does not require a holding/parking device but does require the RCM to be available to perform the function.

The first option was selected as the preferred method to perform the LM exchange operation because the operation could be accomplished solely from the orbiter. Even though a holding/parking device is required, it is a piece of equipment that has been identified for other SOC operations as well as being identified for operations from other space systems studies and, consequently, can be considered as a piece of available equipment. In addition, no requirements are placed on the LM itself and no impacts are identified against the orbiter or the SOC.

Logistics Cradles

The exchange procedure for the logistics cradles envisioned for supporting space construction and flight support operations is essentially identical to that selected for the SOC LM. The HPA holds the cradle for the exchange operation. However, the RCM is required to perform the transport operation because of the distance between the orbiter and the using position of the space construction logistics cradles (Figure 4.5).

In conjunction with the logistics cradle exchange procedure analysis an investigation of the orbiter orientation in relationship to the SOC was performed. The concern was the restrictive path available for the extraction and transport of a logistics cradle from the orbiter to its using position, Figure 4.6. Three orbiter orientations were investigated (1) a 90° rotation, (2) a 45° rotation, and (3) an offset translation. The 90° rotation arrangement, Figure 4.7, was investigated because it simplified the extraction/transport path for the flight support logistics cradle. However, this orientation did not simplify the extraction/transportation function for the space construction logistics cradles. The 45° rotation, Figure 4.8, was a compromise orientation that would simplify this operation. In any of the rotation reorientations of the orbiter, a rotation capability is required at

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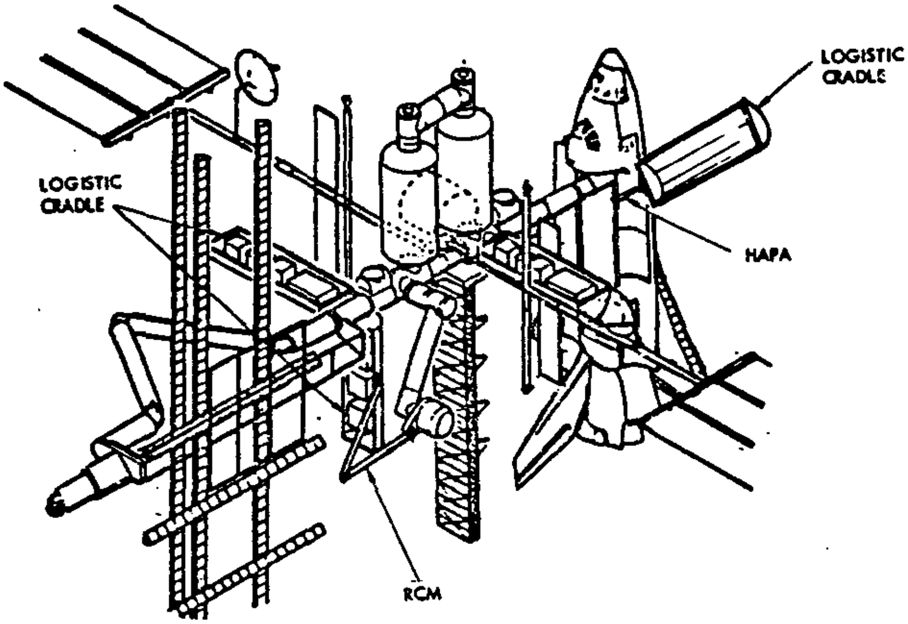

FIGURE 4.5 LOGISTICS CRADLE EXCHANGE PROCEDURE

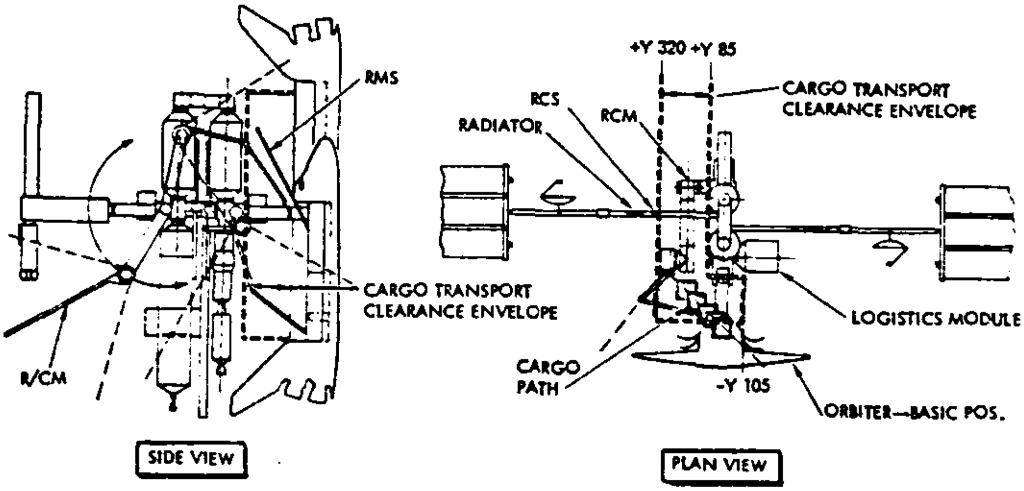

FIGURE 4.6 SOC/ORBITER PAYLOAD EXTRACTION/TRANSPORT CLEARANCE

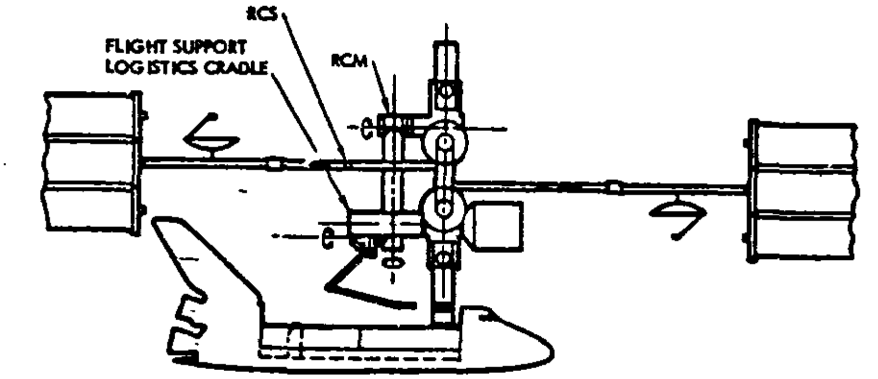

FIGURE 4.7 SOC/ORBITER ORIENTATION CONCEPT - 90° ROTATION

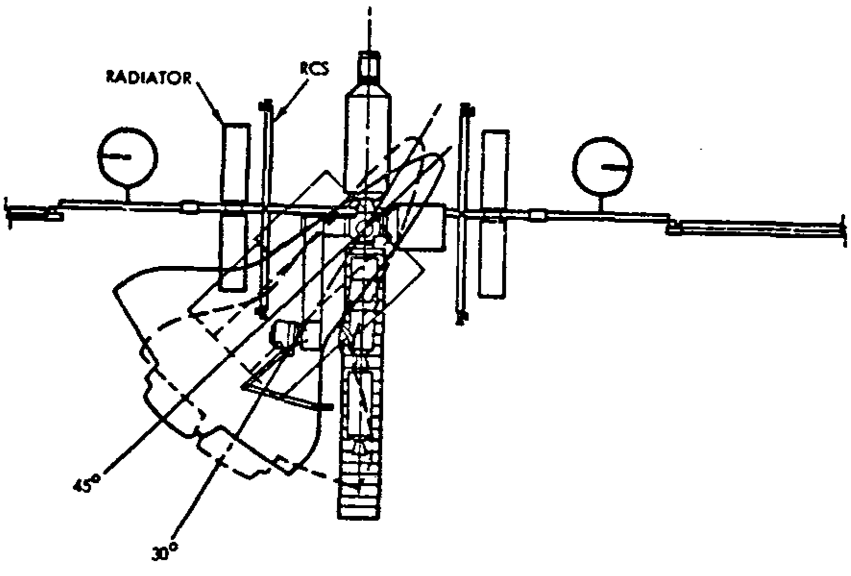

FIGURE 4.8 SOC/ORBITER ORIENTATION CONCEPT - 45° ROTATION

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the orbiter/SOC interface in order to accommodate the utilities interface connections. Figure 4.9 indicates the rotation capability installed on the docking module. The rotations also placed the orbiter in line with the SOC RCS engines, thus subjecting it to the plume impingement contamination. Both the orbiter and the SOC radiators were also partially shaded by these orientations.

The offset translation, Figure 4.10, eliminated the rotation complexities by utilizing the HPA to affect the orientation. However, because the HPA is on the right side of the orbiter, the orbiter must be oriented tail up. This places the orbiter bay clear of the SOC modules and provides a clear path for extraction/transport of the logistics cradle. SOC plume impingement and orbiter radiator blockage, however, are also evident in this arrangement.

However, by utilizing the PIDA device to deploy the payload, the payload is now placed clear of the SOC flight support servicing fixture area and a clear transport path is available, Figure 4.11. Therefore, the normal tail down mated position of the orbiter on to the SOC was retained as the baseline orientation.

4.3 SOC CREW TRANSPORT

The objective of this analysis task was to establish the feasibility of the orbiter to transport a full SOC complement of eight crew members. Implementation of this objective translates into the establishment of a

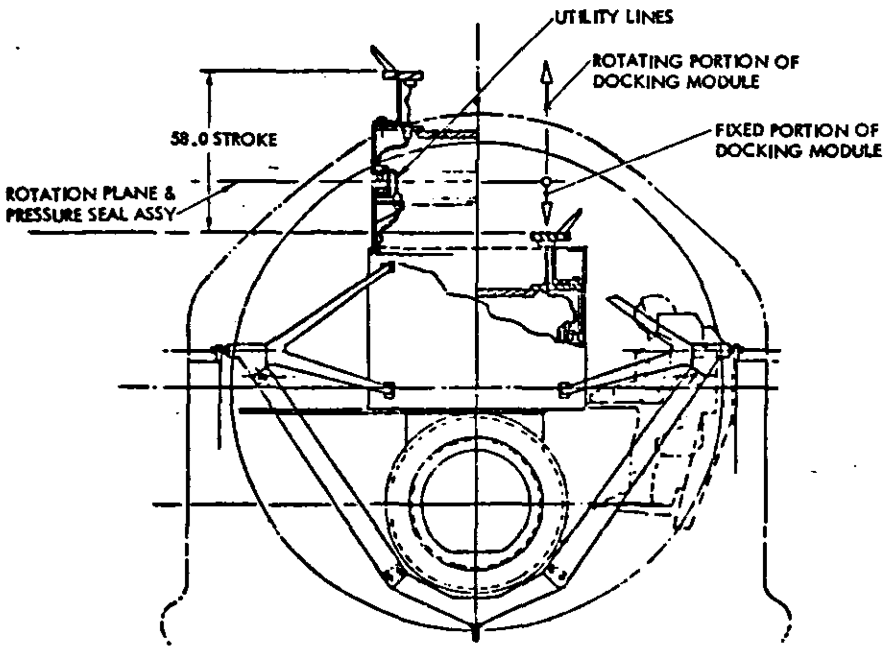

FIGURE 4.9 DOCKING MODULE - ROTATIONAL CONCEPT

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seating arrangement in the orbiter crew cabin that will accommodate a maximum of eight passengers along with adequate storage for the crew emergency support provisions and personal gear, and provide acceptable launch and landing emergency egress paths.

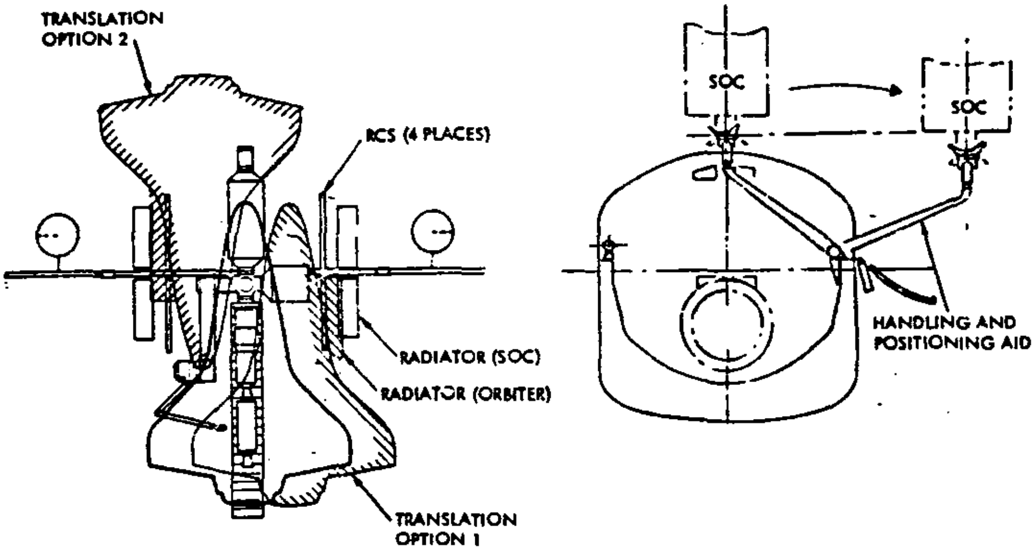

FIGURE 4.10 SOC/ORBITER ORIENTATION CONCEPT - TRANSLATION

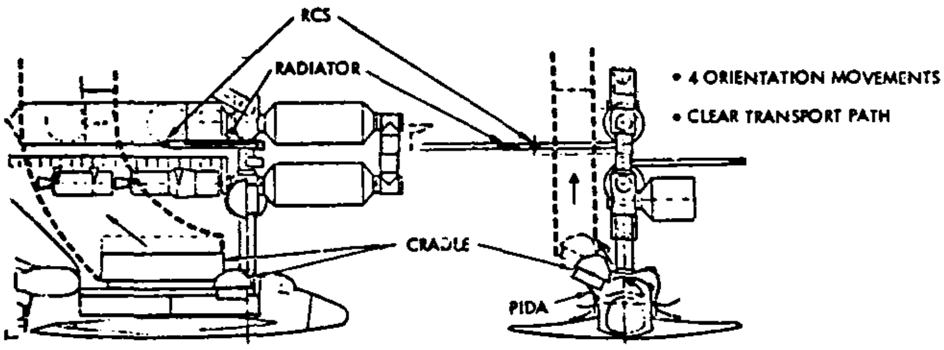

FIGURE 4.11 PIDA DEPLOYMENT/EXCHANGE CONCEPT

The orbiter crew station and the passenger accommodations are comprised of working, living, and equipment storage facilities. Figure 4.12 illustrates the basic orbiter configuration for both the crew flight deck and the mid deck compartments. The flight deck contains the four operational stations of the commander, the pilot, the mission specialist, and the payload specialist. The mid deck contains storage facilities, inflight provisions, and four crew sleep stations. Access to and from the orbiter is through a side hatch into the mid deck area. Access between the mid deck and the flight deck is accomplished by a ladder through an opening in the flight deck floor. Access from the mid deck to the payload bay is via an airlock which is shown within the mid deck. This airlock, however, can be located either in the mid deck or in the payload bay.

Within the basic orbiter cabin three options were analyzed that could accommodate the eight SOC crew members (1) operation of the orbiter by the commander and pilot only, (2) operation of the orbiter with a full four member crew and the airlock within the mid deck, and (3) same as Option 2 except the airlock moved into the payload bay.

If the mission transporting the SOC crew requires only the commander and pilot for its operation, then the two available seats in the flight deck normally occupied by the mission and payload specialists can be utilized as passenger seats. These two seats, in conjunction with the six seats in the mid deck, will accommodate the total eight SOC crew members with no changes to the orbiter, Figure 4.13.

With a full orbiter crew compliment of four, the passenger seating must be accommodated in the mid deck. Two additional seats and passenger accommodation are required. With the airlock located within the crew cabin, the two seats, located as illustrated on Figure 4.14, require the partial removal of the modular storage lockers, Figure 4.15. The remaining storage provisions are indicated and will accommodate the crew emergency equipment and personal gear. The seats are arranged to provide acceptable egress paths for launch pad emergency escape. Access to the galley and waste management areas can be made more accessible by folding and storing the seats adjacent to these areas. Access to L_iOH cannisters and air revitalization equipment will require the folding and storage of three additional seats.

The installation of seat support fittings for the two rescue and emergency equipment are the only changes to the orbiter necessary to accommodate the full SOC crew.

The additional volume gained by removing the airlock permits a more desirable arrangement of the eight seats, Figure 4.16. An area is indicated to accept the storage of the two EVA suits, which are stored in the airlock in the basic cabin arrangement, and to accept the airlock controls. A partition across this area is required to provide a walking surface necessary for launch and emergency egress. The arrangement shown in Figure 4.16 retains the seat arrangement for the three rescue seats and the seat

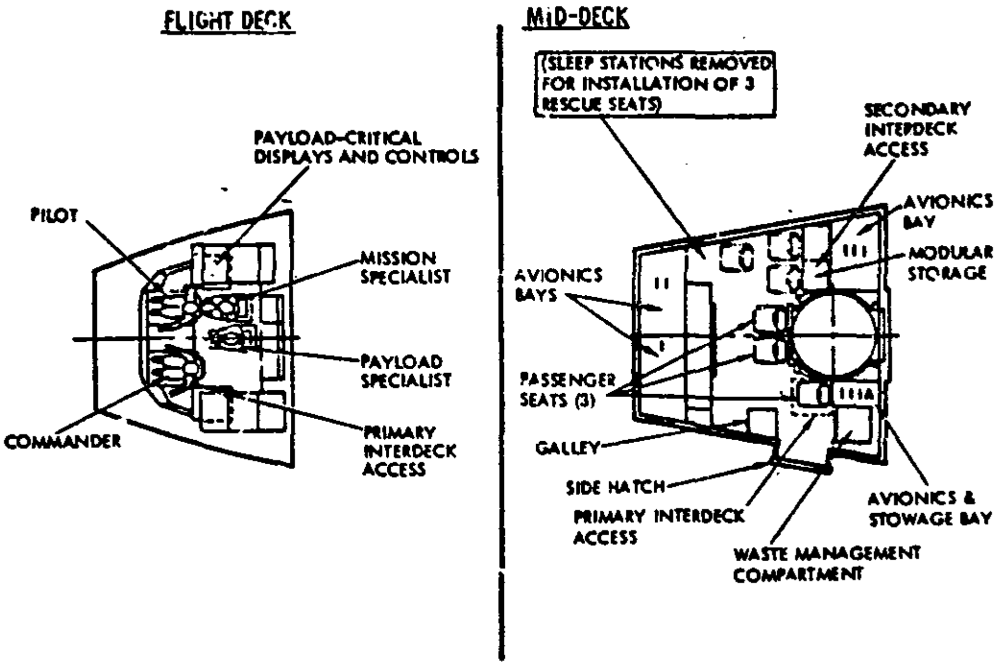

FIGURE 4.12 BASELINE ORBITER CABIN

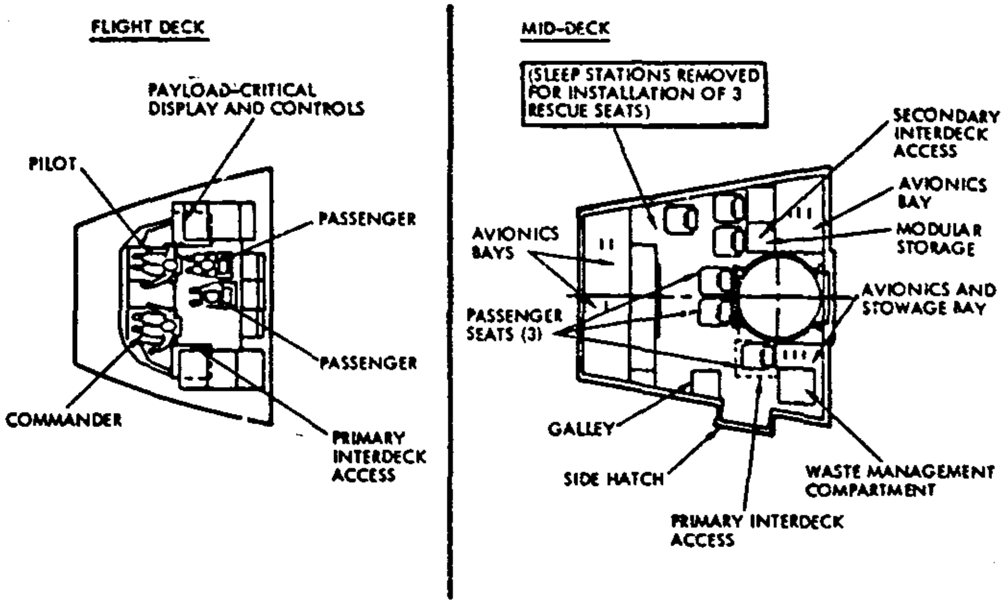

FIGURE 4.13 BASELINE ORBITER WITH 8 PASSENGERS

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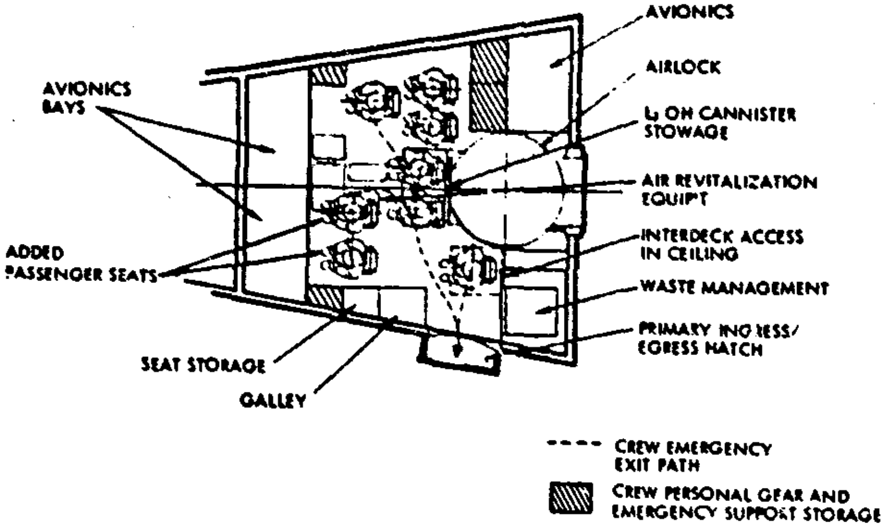

FIGURE 4.14 MID-DECK ARRANGEMENT WITH AIRLOCK

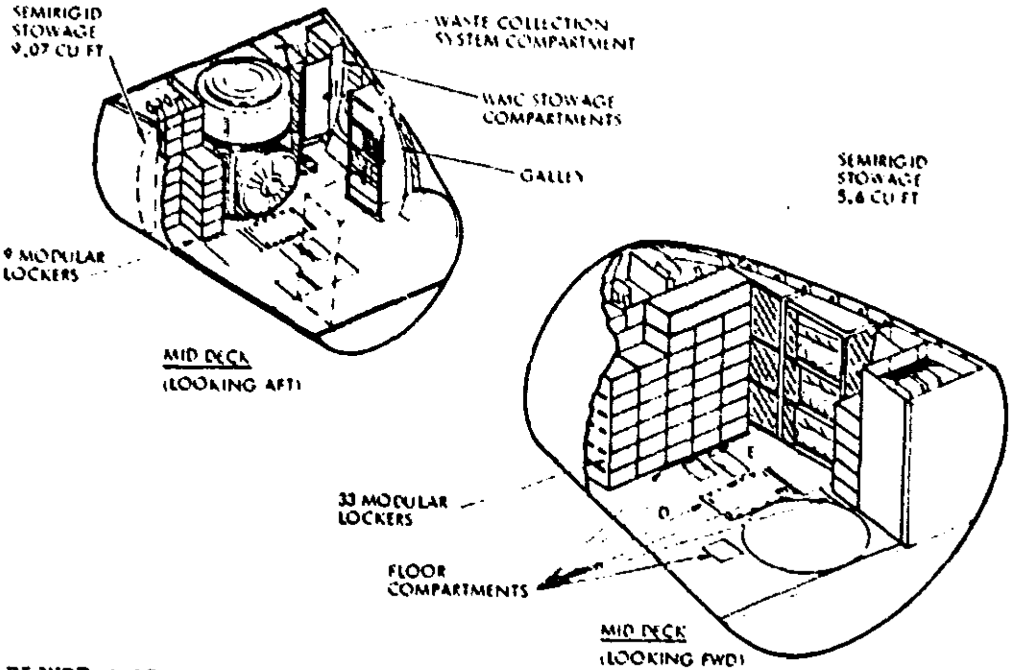

FIGURE 4.15 BASIC OPERATIONAL MID-DECK LAUNCH/ENTRY CONFIGURATION

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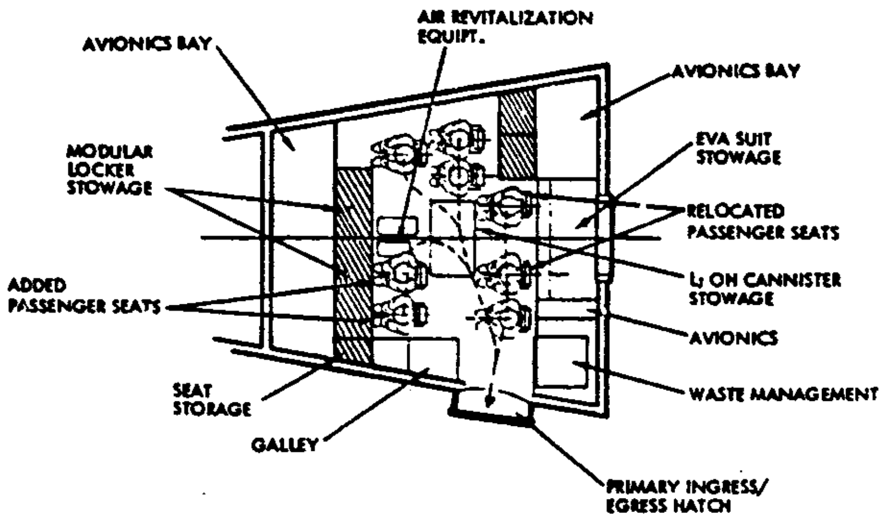

FIGURE 4.16 MID-DECK ARRANGEMENT WITHOUT AIRLOCK

adjacent to the waste management station. By moving the center two seats aft, the location of the two additional seats permits the retention of the modular storage lockers. Acceptable Launch pad emergency egress is provided. Access to the L_iOH canisters and air revitalization equipment is available without folding/stowing any of the seats.

This arrangement, therefore, while being a more desirable arrangement requires the airlock to be moved out of the cabin and seat support provisions added for four seats.

No selected arrangement can be made at this time since the selection is highly dependant on the mission manifest and orbiter cabin arrangement. However, the capability does exist in the Shuttle Orbiter to accommodate eight passengers with or without a full orbiter crew.

4.4 OTV CRYOGENIC PROPELLANT RESUPPLY

The major fluids supplied to SOC are LOX and LH_2 for refueling OTVs based at the flight support facility. Individual OTV flights are expected to require propellant quantities ranging from fractional orbiter loads to as many as three or more orbiter loads per mission. Typical SOC traffic models assume approximately seven orbiter tanking flights per year but 24 or more are possible in later evolutionary phases of SOC.

4.4.1 Basic Approach

The basic concepts and groundrules assumed for resupply of OTV propellants take into account the findings of Reference 1, a preliminary design study of orbital propellant handling and storage, prepered by General Dynamics, Convair Division (GDC).

The basic method adopted for resupply is fluid transfer as opposed to replacement of loaded containers. Although the latter technique does not violate the concept of space-basing, it imposes penalties in terms of OTV inert weight and results in operational complexities. Zero-g fluid transfer has not yet been demonstrated on a large scale, but the basic technology and component designs have been proven in booster and spacecraft propulsion systems and the overall technical risk involved is considered to be very low.

4.4.2 Key Issues and Drivers

The basic issues considered in evaluating the various propellant transfer methods and system design configurations are listed as follows:

- Safety
- Mission Reliability
- Propellant Losses
- Operational Simplicity and Flexibility
- Propellant Positioning
- Zero-G Gaging
- Liquid-Free Venting
- Design Impacts on Shuttle, SOC and OTVs

4.4.3 Baseline System Design

To serve as a reference point for evaluating the various system options, a baseline LOX/LH_2 transfer system was defined using the recommendations of Reference 1. As shown schematically in Figure 4.1, it uses a straightforward, low-risk, conservative approach. The main feature of the baseline system are:

- Nested, dewar type LOX/LH_2 supply tanks in the orbiter
- Full screen-channel capillary propellant acquisition system in the supply tanks
- Helium pressurization for propellant transfer

- Press/vent cycles for prechilling the LH_2 receiver tanks in MOTV

- Liquid separators in the OTV vent lines

- RF (radio frequency) zero-g guaging of tanked propellant quantity

Prechilling of the LH_2 receiver tank is accomplished by alternately spraying small amounts of liquid into the tank, allowing it to vaporize on the warm walls and venting the boiloff vapors as required to prevent overpressurization. After approximately three of these pressure/vent cycles, the tank is sufficiently cold to accept the full load of liquid without further venting, aided by mixing jets in the tank which help condense the excess ullage vapors. Pressure/vent cycling is not required for the LOX receiver tank because of the larger sensible heat capacity per unit volume of oxygen. Termination of tanking is based on indication of a full condition by the RF zero-g guaging system whose sensing antenna is shown installed in LH_2 tank of Figure 4.1.

The vacuum jacketed, nested design of the orbiter supply tank is relatively heavy but simplifies ground operations by eliminating the need for purging MLI (multilayer insulation) while on the Shuttle launch pad. Although not shown, a venturi device is provided to saturate the helium pressurization gas flow with propellant vapor, so as to avoid dry-out of the capillary screens in the supply tank. Figure 4.17 presents an estimate of the typical propellant losses incurred between Shuttle launch and MOTV final usage. Of the 7.3% overall loss, 3.6% is chargeable to the process of propellant resupply.

4.4.4 Safety and Reliability Considerations

Propellant handling and refueling are generally considered one of the more hazardous operations encountered in ground and orbital operations. The actual risk, however, can be reduced to very low levels if proper design philosophy and operating procedures are used. NASA Handbook 1700.7, "Safety Policy and Requirements for Payloads Using the Space Shuttle Transportation System" (Reference 4) sets forth design and operational rules that also pertain to propellant transfer systems in general.

Following is a list of the more important safety guidelines and considerations applicable to the SOC propellant resupply:

- <u>Redundancy</u> - No single mechanical or electrical failure will endanger personnel or equipment. As a design goal, no single mechanical or electrical failure (except loss of power) will preclude the transfer of propellant.

- <u>Leakage</u> - As a design goal, three independent mechanical inhibits will be provided to prevent leakage or unplanned discharge of propellant into the orbiter cargo bay or other

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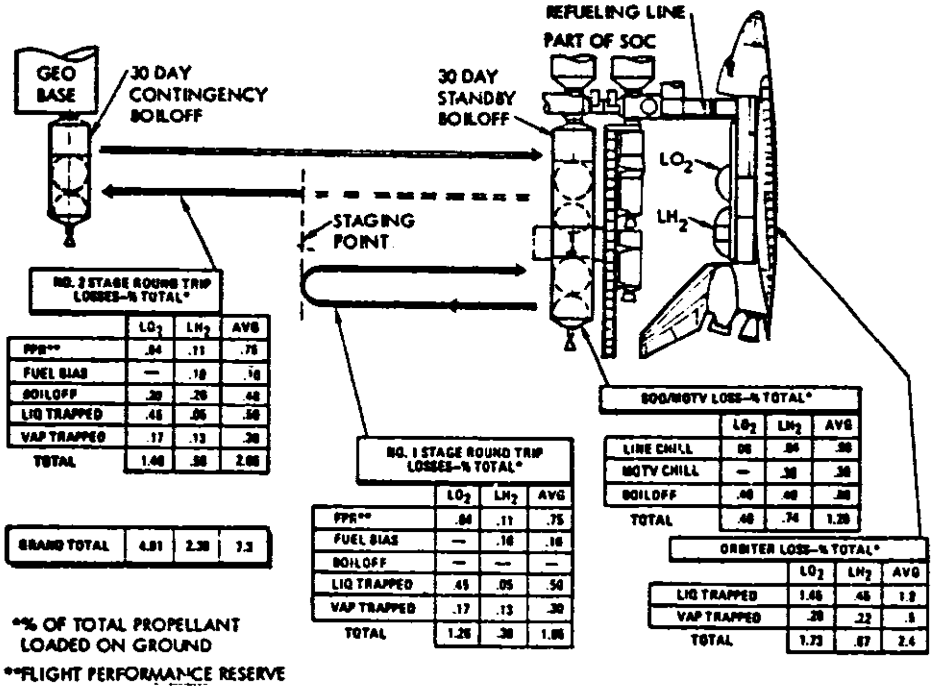

NO. 2 STAGE ROUND TRIP LOSSES—% TOTAL*

	LO_2	LH_2	AVG
FPR**	.64	.11	.75
FUEL BIAS	—	.16	.16
BOILOFF	.20	.28	.48
LIQ TRAPPED	.45	.05	.50
VAP TRAPPED	.17	.13	.30
TOTAL	1.46	.58	2.05

GRAND TOTAL	4.81	2.38	7.2

NO. 1 STAGE ROUND TRIP LOSSES—% TOTAL*

	LO_2	LH_2	AVG
FPR**	.64	.11	.75
FUEL BIAS	—	.16	.16
BOILOFF	—	—	—
LIQ TRAPPED	.45	.05	.50
VAP TRAPPED	.17	.13	.30
TOTAL	1.26	.38	1.65

SOC/MOTV LOSS—% TOTAL*

	LO_2	LH_2	AVG
LINE CHILL	.06	.06	.06
MOTV CHILL	—	.30	.30
BOILOFF	.40	.40	.80
TOTAL	.46	.74	1.20

ORBITER LOSS—% TOTAL*

	LO_2	LH_2	AVG
LIQ TRAPPED	1.45	.45	1.9
VAP TRAPPED	.28	.22	.5
TOTAL	1.73	.67	2.4

FIGURE 4.17 BASELINE CRYO PROPELLANT LOSS MODEL

enclosed areas. Quick-disconnect fittings should include means for verification of interface seal integrity by inert gas leakage checks before exposure to propellants. Means should be provided for detecting external leakage from the disconnect during use, as well as from other system components. Purging capability should be provided for such contingencies.

- Monitoring - Critical system measurements should be displayed and audible warning and lights provided to indicate out-of-tolerance conditions. Critical valves shall have position indicators, and automated override circuitry shall be provided to automatically correct hazardous conditions through the proper sequencing of electrical commands.

- Line Routing - Propellant lines, valves, equipment and other potential sources of leakage should be located outside of enclosed spaces, and protected from inadvertant physical damage. LOX and LH_2 plumbing and tankage should be widely separated wherever practical.

- <u>Procedures</u> - Formal detailed operating procedures, check lists, diagnostic routines, and contingency plans should be prepared and followed at all times.

- <u>Abort Landing</u> - Propellant tankage installed in the orbiter shall be capable of landing safely with propellant contained, under the design load factors specified for emergency or abort landing in NHB 1700.7; or means shall be provided for safely dumping the propellants over board prior to such landing. The propellant system and tankage should be able to safely withstand the soak-back temperature peak experienced in the cargo bay after landing at a contingency field not equipped with a mobile air purging unit.

4.4.5 Advanced System Concepts

. A wide range of system concepts were evaluated, including design options presented in References 1 and 2. Figure 4.18 shows some of the more promising candidate refinements integrated into a propellant transfer system having reduced boiloff losses and improved operating features. The following combines a description of system operation with an evaluation of the candidate refinements.

<u>System Installation</u>

Figure 4.19 shows an example scenario with an orbiter propellant resupply tanker docked to SOC, the configuration of the flight support facility, and the routing of the interconnecting propellant transfer lines. An RCM is shown in the process of connecting a transfer line to an MOTV by means of a quick-disconnect fitting. An RCM is also used to engage a SOC-mounted folding boom and quick-disconnect with the orbiter resupply tank. A shirtsleeve environment refueling control center (not shown) is located on SOC, along with other system requirements such as transfer plumbing, pumps and storage space. This arrangement minimizes the system inert weight that must be carried on the orbiter. Wherever possible, lines, valves, pumps and mechanical equipment on SOC are mounted externally for accessibility, and to minimize leakage of any fluids or gases into enclosed areas.

<u>Refueling Control</u>

The main console and major control/monitoring functions for propellant transfer are located in the shirtsleeve - environment of the SOC refueling control center which is positioned to afford good visibility of the major propellant system elements and to monitor or backup the RCM operator's cab during handling of the transfer flex lines and engagement of the propellant disconnects. Refueling operations are remotely controlled and automated where practical. No EVA is required except in cases of equipment malfunction, in which event provisions are made in the design of the equipment for EVA maintenance and/or backup operation using standardized EVA tools.

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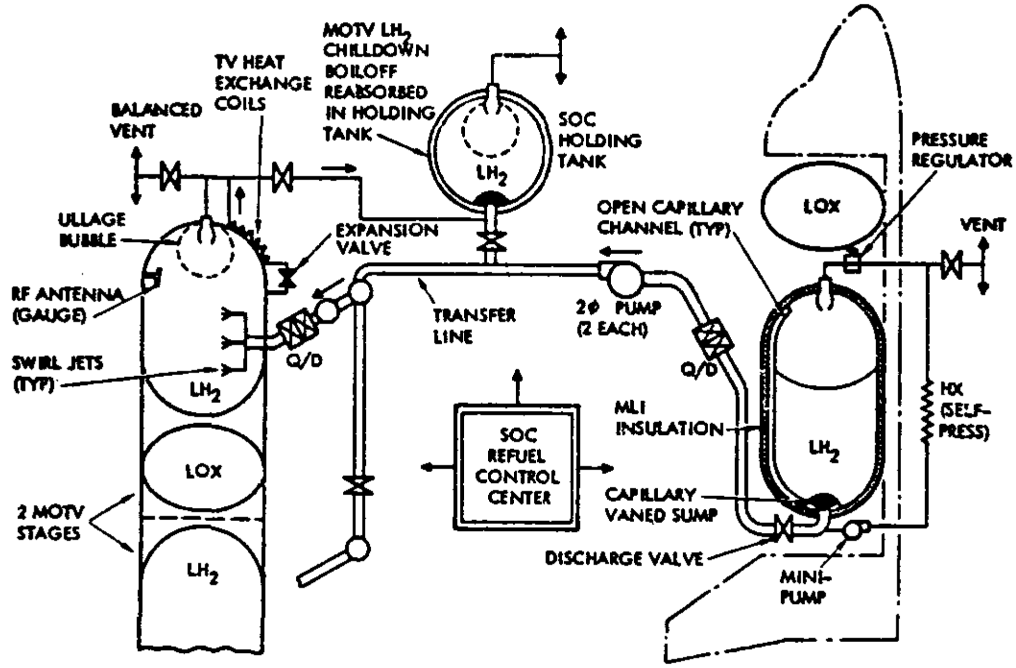

FIGURE 4.18 ROCKWELL SOC REFUELING SCHEMATIC

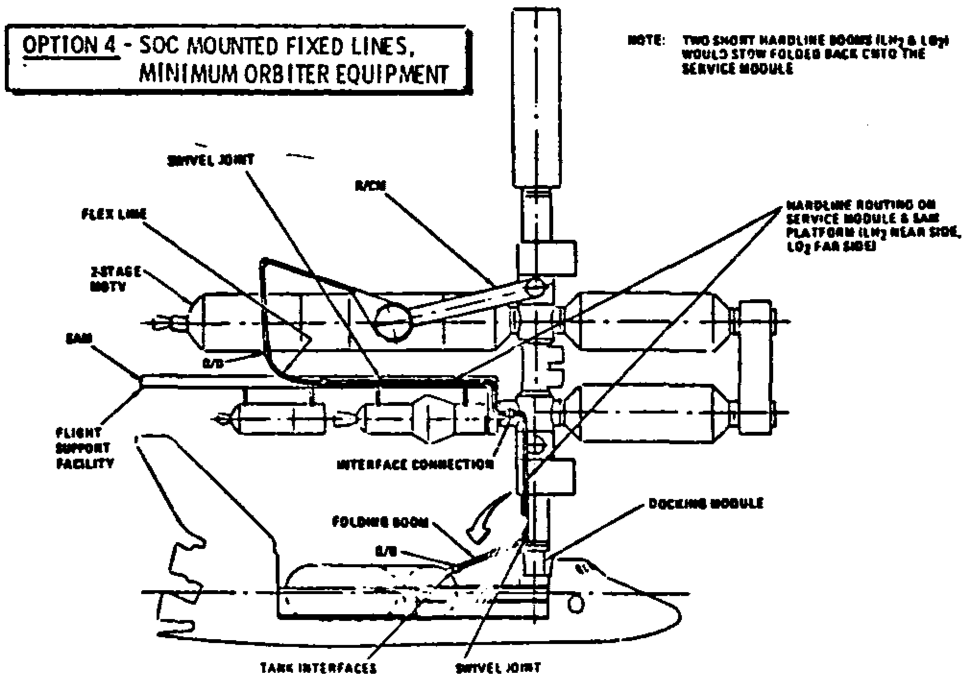

FIGURE 4.19 FUEL TRANSFER LINE CONCEPT

The above operating philosophy minimizes exposure of personnel to EVA hazards, simplifies operations and reduces the time required for refueling. It is recommended that a minimum backup and emergency override control/monitoring capability be provided in the orbiter using the aft flight deck and/or co-pilot station. The senior refueling specialists would be members of the SOC crew, with orbiter personnel receiving minimum training as required for safety monitoring and emergency override functions.

Supply Tank Design

Instead of the baseline nested dewar configuration, the orbiter propellant supply tanks shown in Figure 4.18 are un-nested and insulated with MLI to minimize system inert weight. A five mil Mylar scuff layer or purge bag encloses the MLI to permit purging with GHe on the Shuttle launch pad, in order to avoid liquifaction of GN_2 on the outer surface of the LH_2 tank wall. Moisture condensation on the outside of the purge bag will be prevented by purging the orbiter cargo bay with dry GN2. A 1-to-2 inch layer of foam insulation sprayed on the tank wall (underneath the MLI) is recommended to reduce boiloff rates during launch countdown. The purge bag and MLI layers are perforated to allow rapid venting and outgassing of GHe during boost and orbital flight. The purge-bag/MLI/foam combination also provides an adequate level of meteorite protection for the tanks.

The combined installed length of the un-nested LOX and LH_2 supply tanks is approximately 7.9 M (26 feet), based on a total tankage system weight of 29,500 Kg (65K lb) full. If the LOX tank is forward as shown in Figure 4.18, the tank combination can be installed flush against the rear bulkhead of the cargo bay with its c.g. well inside the prescribed limits. This provides the greatest flexibility in cargo manifesting; e.g., if the supply tanks are not fully loaded, additional cargo such as construction materials will have an uninterrupted forward bay length of 10 M (34 feet) for storage. With the LOX tank aft as in Reference 1, the remaining bay length would have to be divided between the forward and aft ends [4 M (14 feet) aft and 6 M (20 feet) forward] to meet c.g. requirements. Anti-slosh ring baffles (not shown) would be installed in the tanks to control propellant dynamics during Shuttle boost with either full or partial propellant loads. It should be noted that the tank lengths shown in Figure 4.18 are exaggerated for schematic clarity.

Subcooling of Propellants

In the baseline system, the LOX and LH_2 propellants would slowly boil in the supply tanks during countdown at a vent system back pressure of about 1.1×10^5 N/M^2 (16 psia), corresponding to saturation temperatures of 85.6°K (164°R) for LOX and 20.5°K (37°R) for LH_2. These temperatures can be lowered close to the triple point (freezing) temperatures of 54.4°K (98°R) for LOX and 13.8°K (24.8°R) for LH_2 by the simple technique of bubbling helium gas through the bulk propellant and out through the launch facility vent system. Subcooling has the potential of greatly reducing propellant boiloff losses in orbit.

Helium injection is commonly used in boost vehicles for reducing cryogenic propellant temperatures and rocket pump NPSH requirements. This method of subcooling is based on the law of partial pressures and is identical to the process of evaporative cooling produced by flowing dry air over a wetted surface.

Substantial subcooling of the propellants reduces their vapor pressures below atmospheric, with the balance of the tank absolute pressure (16 psia) made up by the partial pressure of helium in the ullage. This prevents negative pressure differentials across the tank walls. If desired, this helium-rich ullage can be vented overboard in orbit, prior to the start of propellant transfer.

By increasing the rate of helium injection in the tanks it is even possible to produce a solid/liquid mixture or "slush" at the triple-point temperatures. This mixture has a greater thermal capacity because of its heat of fusion, and would delay boiloff in-orbit longer than would pure subcooled liquid. As indicated in Reference 3, slush mixtures have been pumped and transferred in the same manner as pure liquids, however, more demonstration testing is needed to verify the feasibility of using slush for propellant resupply.

Autogenous Pressurization

To avoid the weight and logistics requirements of a helium pressurization system, an autogenous system or self-pressurization loop can be used as shown in Figure 4.18. Liquid propellant is vaporized by an ambient heat exchanger coil which pressurizes the tank ullage in "bootstrap" fashion. The pump head requirement can be very low (just enough to overcome fluid friction losses in the flow loop). Conceivably, the pump could be replaced by a capillary wicking device which transports liquid to the heat exchanger just as in a conventional heat pipe.

Pumped Transfer

If a pump is provided to transfer propellant to the OTVs, pressurization of the supply tank can be reduced, or even eliminated if a zero-NPSH (net positive suction head) pump design is provided that is capable of handling two-phase boiling liquid. Such devices (known as boost-pumps) have been developed for use in LOX/LH_2 propulsion systems and are available off-the-shelf. If such a device is used, no external pressurization system is required for the supply tank, which can self-pressurize by flashing of the bulk propellant as it is withdrawn. The electrical power required for pumped transfer of 29500 Kg (65,000 lb) of LOX/LH_2 at 6:1 mass ratio is on the order of 350 watts for a sequential transfer in four hours.

Propellant Acquisition

The screen-channel type of capillary propellant positioning system, recommended by Reference 1 and shown in the baseline system of Figure 4.1, is commonly used for propellant acquisition, especially where high adverse "g" levels are encountered as in the orbiter OMS and RCS tanks. Screen channels, however, require special filling procedures, are subject to bubble trapping and vibration failures, and are expensive to develop and fabricate. The simplified capillary acquisition system shown in Figure 4.18 consists of open wicking channels along the walls and a cluster of vanes at the sump, as shown in greater detail by Figure 4.20. This design is an outgrowth of the capillary system design used successfully in the Viking Orbiter Spacecraft tank shown in Figure 4.21. The expulsion efficiency attained by this design in actual service has been estimated at 99.5%. Although that system was designed for lateral disturbances up to only 10^{-4}g, much higher adverse disturbances can be accommodated by using a finer grained arrangement of capillary surfaces in the sump area as shown in Figure 4.20. The maximum disturbance during SOC propellant transfer are expected to be on the order of 10^{-3}gs. For that application the simple vane-type design is considered superior to a screen-channel acquisition system in terms of cost, reliability and operational simplicity.

Liquid-Free Venting

The simplified vane-type capillary system design described above can also be used in the OTV tanks and the SOC storage tanks. As shown in Figure 4.18, capillary forces are produced which displace the vapor bubbles in the tanks to form a single ullage space surrounding the vent tube. Also, a slow rotation or swirling action in the bulk liquid can be provided by recirculation pumps or incoming transfer flow to create centrifugal "g" forces, buoyancy effects and radial/axial circulation patterns which dislodge bubbles from the tank walls and sweep them to the vent port, thereby positioning the ullage much more positively at its desired location. Normal boiloff can then be handled through the main vent system without the need for a liquid separator. The other options for achieveing liquid-free venting are mechanical separators or the use of a thermodynamic vent (TV) system shown schematically on an OTV tank in Figure 4.18. Liquid from the expansion valve evaporates in heat exchange coils attached to the tank wall, thereby cooling the wall and intercepting heat leaking through the insulation. A TV system, however, adds system weight and can be costly to fabricate.

Chilldown and Transfer Procedures

Assuming that an LH_2 storage or holding tank exists on SOC as shown in Figure 4.18, boiloff vapors from chilldown of the transfer line and receiver tank would not be vented overboard as with the baseline transfer system. Instead, they would be bubbled into the bulk liquid of the SOC storage tank where they would be condensed by direct contact and recovered. When the OTV receiver tank is sufficiently chilled, the vent line to the storage tank would be closed and filling of the OTV tank completed with the

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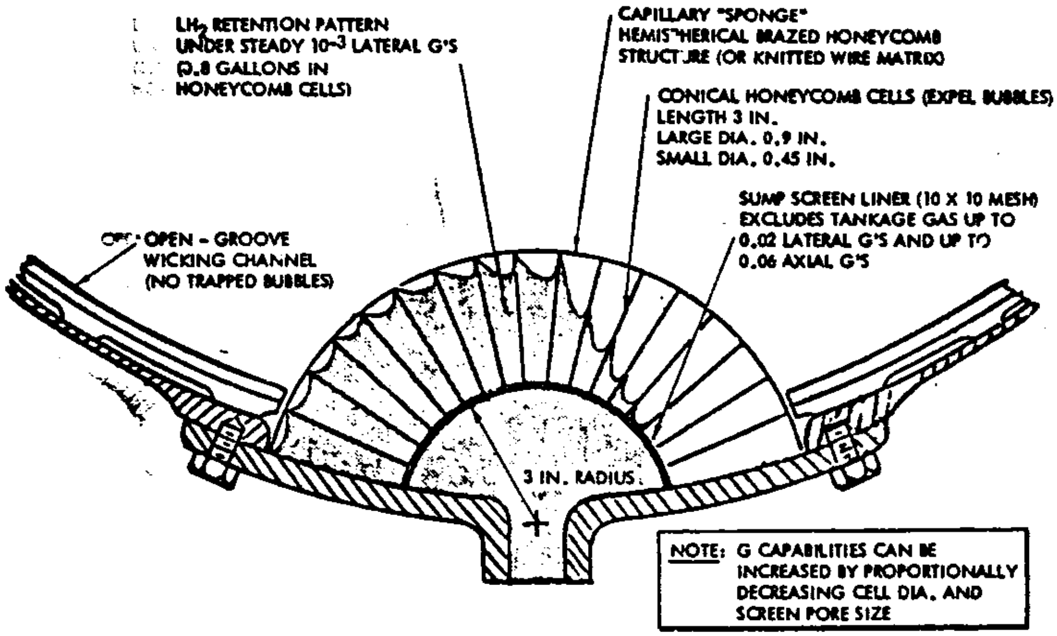

FIGURE 4.20 CAPILLARY MATRIX TYPE SUMP

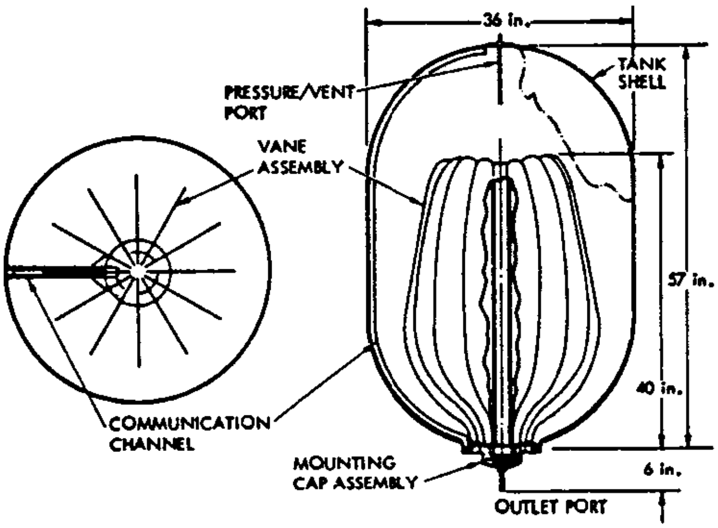

FIGURE 4.21 VIKING ORBITER TANK AND PROPELLANT MANAGEMENT DEVICE

aid of jets that provide bulk mixing within the tank to condense the displaced ullage vapor. With this approach, pressure/vent cycling during chilldown is not required. It is not necessary for the receiver tank to be evacuated to establish pump chilldown and priming, if sufficient pressure is provided in the orbiter tank to drive the receiver tank ullage and chilldown vapors into the SOC storage tank for condensation.

Another design option is the use of a vapor return line (not shown) between the receiver tank and supply tank ullages, creating a closed loop flow system. As such the two tanks are always at the same pressure except for flow friction losses in the vapor return line. Under zero-g conditions, with all valves in the flow loop open, pump chilldown and priming could be initiated with just the slight head produced across the transfer pump by spinning it in the presence of propellant vapor. Excess boiloff would either be admitted to the SOC storage tank for condensation, or absorbed back in the orbiter supply tank with the aid of special vapor nozzles located beneath the liquid surface to allow condensation of the returning vapor by direct contact.

Another version of the vapor return technique uses a fan or blower in the vapor return line, instead of a pump in the liquid transfer line, to provide the pressure differential needed for transfer (Reference 2).

Another version uses a "cold trap" or condensing coil in the sump of the supply tank to recover chilldown boiloff, as shown in Figure 4.22.

Recovery of chilldown boiloff in the supply tank liquid can also be accomplished without a vapor return line by use of a flow reversal method as follows: With an ullage pressure established in the supply tank that is greater than in the receiver tank, the valve at the supply tank liquid outlet is opened long enough to admit a metered slug of liquid in the warm transfer line where it is tapped, vaporizes and builds up a pressure in the receiver tank which is somewhat higher than in the supply tank. Opening the valve again at the supply tank outlet will permit reverse flow of this vapor through the liquid bulk where it is recondensed. After pressures equalize, the supply tank outlet valve is closed, the supply tank pressure is again increased (by the pressurization system) and the cycle is repeated. through successive cycling, the transfer line and receiver tank are progressively cooled until chilldown is complete and propellant transfer can begin. Because this method is somewhat complex, it would be used only in case the SOC storage tank did not contain enough liquid to absorb the chilldown boiloff vapors without exceeding tank design pressures.

Reduction of Propellant Losses

By use of the system options described above, propellant losses chargeable to propellant resupply can be reduced from the 3.6% shown in Figure 4.17 for the baseline system to less than 0.25% if recovery of chilldown boiloff is used and the supply tank is warmed to minimize its tapped ullage mass. The 0.4% OTV mission boiloff loss can be eliminated by last-day fueling with subcooled propellant from a SOC storage tank.

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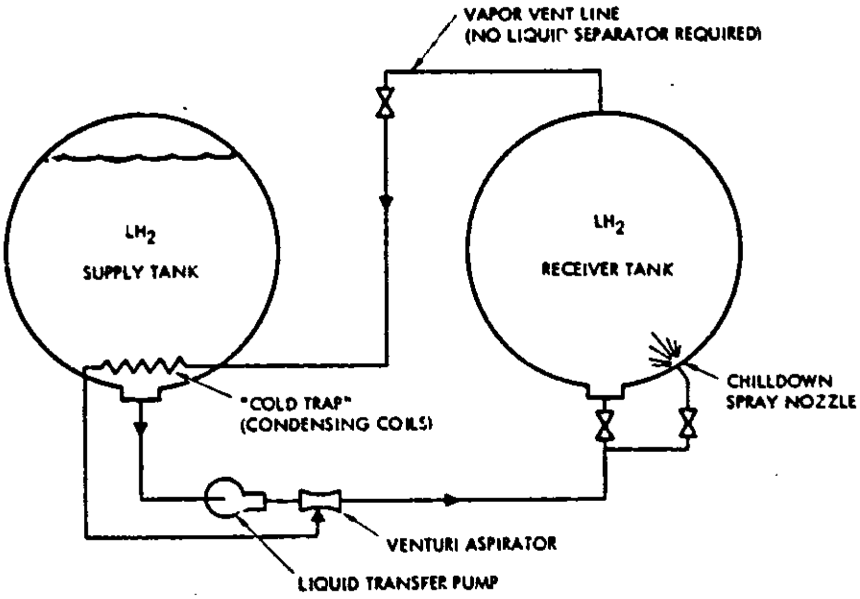

FIGURE 4.22 ZERO-LOSS LH_2 RECEIVER CHILLDOWN CONCEPT

Active Refrigeration

The concept of active refrigeration on SOC for cryogenic propellants provides the potential for reducing standby boiloff losses and/or insulation requirements for the OTVs and/or SOC propellant storage tanks. The best current technology for spaceborne cryogenic refrigerator systems is typified by the mini-Halo design shown in Figure 4.23. In principle, it is a reversed, closed-loop Brayton power cycle using helium as the working fluid. Although the expansion ratio across the turbine is very low and can cool the helium flow only a few degrees by itself, cryogenic temperatures are achieved at the cooling load by a bootstrap process in the regenerative heat exchanger which precools the helium entering the turbine. In cases where more than one cooling temperature is required, additional expansion turbines flowing in parallel are installed across the two legs of the regenerative heat exchanger at the appropriate temperature points. Theoretically, unlimited refrigerator life can be provided in the hardware design by means of magnetic or gas dynamic bearings for the compressor, turbine and electrical motor drive.

Figure 4.24 shows a refrigerated shield/insulation concept suitable for an LH_2 storage or OTV tank. Although the cooling tubes could handle the cold helium flow directly from the expansion turbine, the added system pressure drop in the helium flow loop would seriously degrade refrigerator performance. A better arrangement would be the use of an intermediate heat

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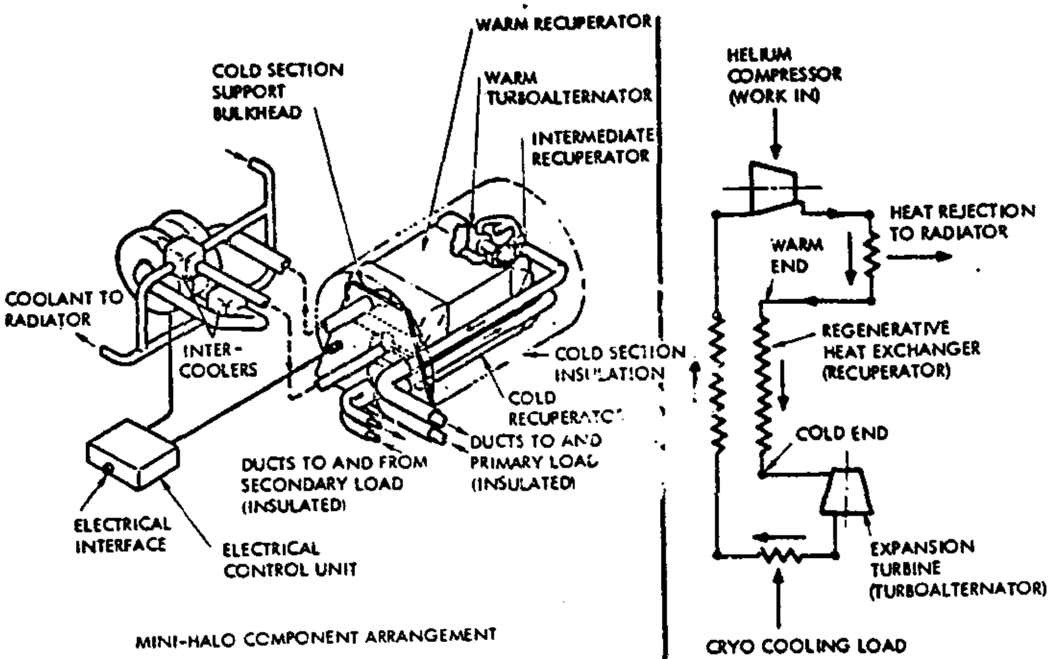

FIGURE 4.23 BRAYTON TURBO REFRIGERATOR SCHEMATIC

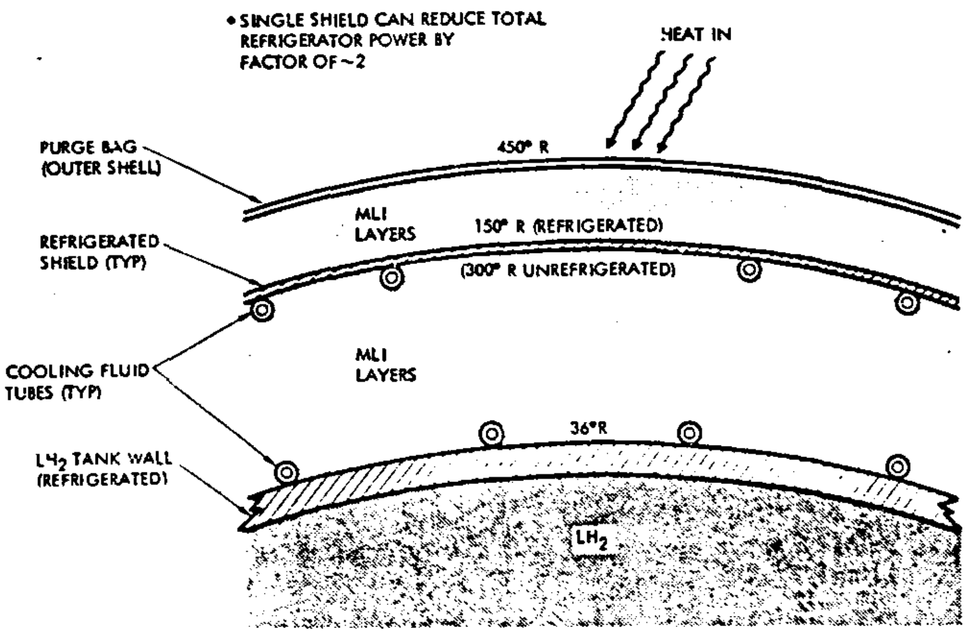

FIGURE 4.24 LH_2 REFRIGERATED SHIELD INSULATION CONCEPT

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transfer fluid pumped in a closed loop through the tank coils and a heat exchanger at the refrigerator cooling interface. For 36°R, a good candidate fluid would be two-phase hydrogen, boiling in the tank coils and condensing in the interface heat exchanger. At 150°R, a good candidate fluid would be two-phase nitrogen.

The refrigerated shield shown in the center of the MLI blanket of Figure 4.24 is optional, but it reduces the power requirement of the refrigerator by a factor of approximately two as shown in the curves of Figure 4.25. The increased power requirement shown for a typical OTV stage versus a storage tank is due to the thinner insulation blanket and larger structural heat shorts inherent in OTVs which must transmit engine thrust loads into the tanks.

As shown in Figure 4.25, the refrigeration power requirements are moderate even for the higher OTV heat load. It can also be seen that refrigeration of LOX is much less important than for LH_2 if subcooled propellant is provided by the orbiter. If SOC storage tanks are provided, there is even a serious question whether refrigeration would be cost effective in cases where subcooled propellants are used. Since a large subcooled tank of LH_2 could last 500 days before venting is requied, normal usage and refilling with subcooled LH_2 on a 180 day turnover cycle, should always keep the storage temperature well below boiling for a 16 psia venting pressure.

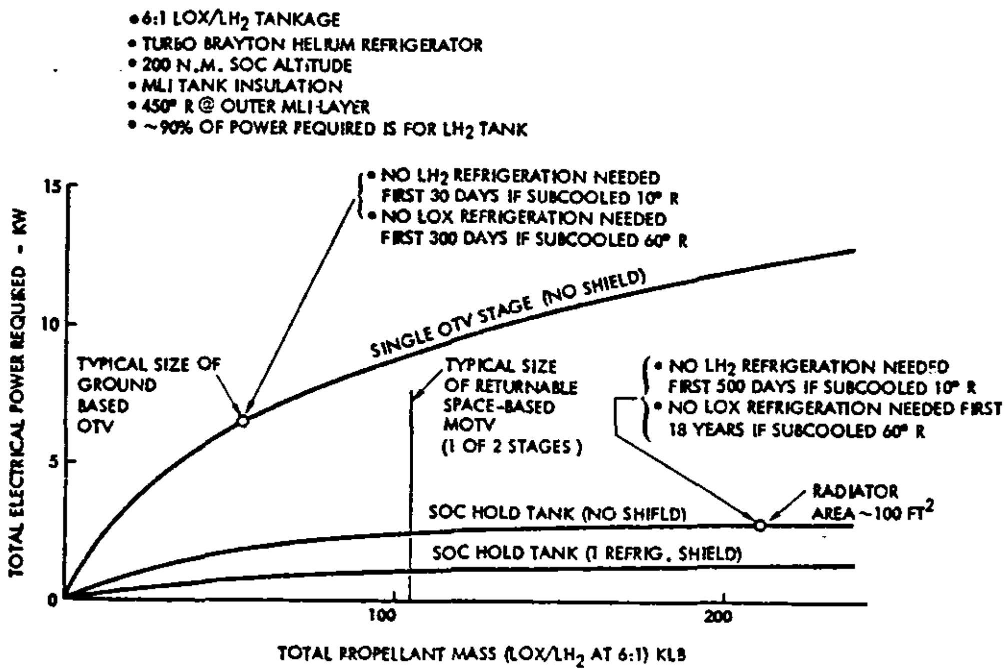

FIGURE 4.25 CRYO PROPELLANT REFRIGERATION POWER

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If refrigeration was provided, however, propellant delivered to the OTVs could be subcooled to a greater degree, perhaps reducing OTV mission boiloff and/or insulation requirements. Further study of OTV traffic requirements is needed in order to evaluate the cost-effectiveness of active refrigeration.

4.5 OTV PROPELLANT STORAGE AND SYNERGISTIC INTERACTIONS WITH SOC

Incorporation of an OTV propellant depot function on SOC can provide major benefits in terms of logistical flexibility and substantial reduction in the required number of Shuttle launches. These synergistic interactions are diagrammed in Figure 4.26 and discussed in the following subsections.

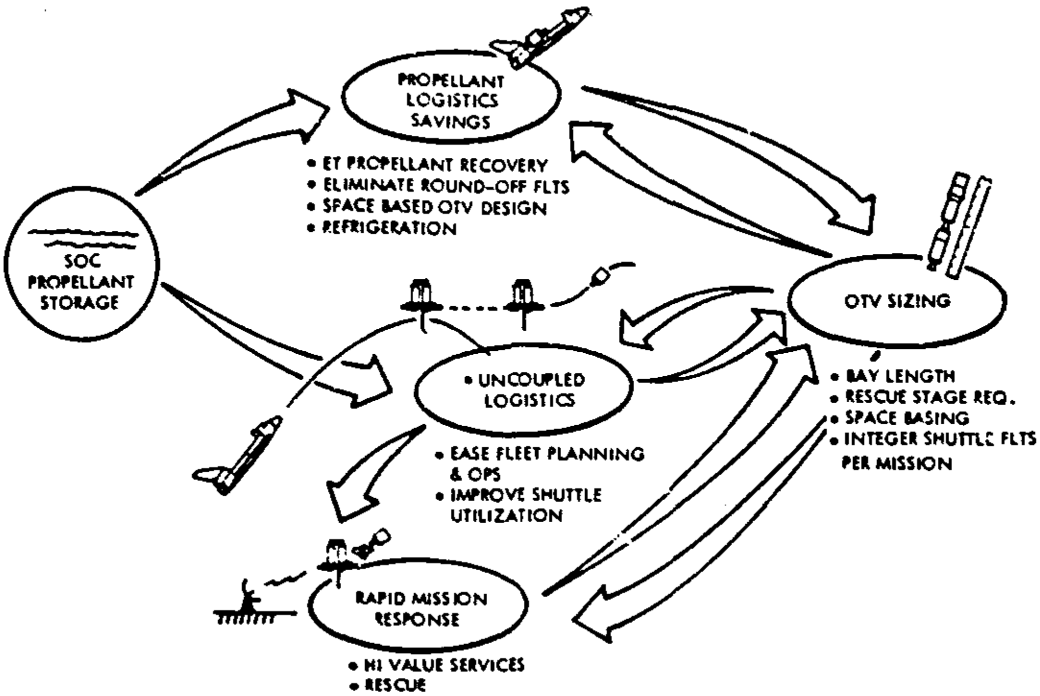

FIGURE 4.26 SOC PROPELLANT STORAGE - SYNERGISTIC INTERACTIONS

4.5.1 Orbiter Utilization

SOC storage of OTV propellant permits better usage of orbiter cargo capacity by "piggybacking" LOX and LH_2 on orbiter flights which are either underloaded or volume limited as shown in Figure 4.18. This propellant would be accumulated in well insulated holding tanks on SOC until needed for refueling. Typical cargo weights on non-tanking logistics flights are on the order of 13,000 Kg (30,000 lb), leaving 16,000 Kg (35,000 lb) excess capacity for propellant fill-in. In the case of LOX, this weight can be carried in a specially designed container occupying only six feet of cargo bay length, while an LH_2 tank to carry that weight would occupy about 15.5 M (51 feet) of bay length. For LOX/LH_2 in a 6:1 mass ratio, combined length of the two un-nested tanks would be approximately 4.9 M (16 feet). With low density, volume-limited cargos such as construction materials, some of this cargo would be displaced by the propellant container(s) but could be fitted into later flights with proper prescheduling of SOC operations to the benefit of overall logistics efficiency.

Depending on the actual logistics traffic characteristics, as many as one-third of the total Shuttle launches could be saved by propellant storage, alone. Additional savings are possible through sub-orbital recovery of propellant residuals from the Shuttle ET (external tank). As described in the next section, these techniques combined with space-basing of OTVs can save up to one-half of the Shuttle launches which might otherwise be required.

4.5.2 Uncoupled Logistics

Propellant storage on SOC greatly increases the number of payload mixing options open to the Shuttle cargo planner and subsequently improves the flexibility of STS operations by decoupling orbiter logistics from SOC based OTV operations. Instead of having to break into the Shuttle manifesting plan for a cluster of three, four, or possibly five closely spaced flights in support of an MOTV mission, the propellant could be delivered to SOC on a routine scheduled basis, thus, easing fleet management and potentially improving fleet utilization. It also eliminates any round-off flights needed above an integer number of Shuttle launches for a given OTV mission. The amount of propellant required for each OTV mission can vary greatly depending upon the amount of payload to be carried and the mission destination. It is further influenced by OTV size and performance characteristics. Thus, the propellant actually required in support of the various missions will frequently require a fractional Shuttle load of propellants, or round-off flight to match their varying needs. Over the long term this would average to one-half of a Shuttle flight per OTV mission. This extra one-half Shuttle flight per mission could be eliminated with propellant storage on the SOC.

4.5.3 Reduction of Boiloff Losses

Because the weight or mass fraction of a SOC mounted storage tank is not critical, as it is for flight hardware on an OTV, the performance of its thermal insulation can be much higher through use of additional layers of MLI and sophisticated insulation techniques. These include vapor-cooled shields which intercept incoming heat/leakage with the sensible enthalphy of the boiloff vapors. Also, the surface-to-volume ratio of storage tanks can be more favorable than for OTV tanks by virtue of their larger size.

If subcooling of delivered propellants is used, boiloff of LH_2 in a large storage tank can be delayed for as long as 500 days compared to 30 days maximum for typical OTV flight weight tank systems. Also, boiloff during OTV standby can be eliminated since missions requiring more than one orbiter load of propellant can initiate all tanking within a day or so prior to OTV departure. Boiloff during a long OTV sortie can also be reduced by using the greater initial subcooling of propellant at the start of the mission. Also, SOC storage can provide a convenient thermal sink for recovery of childown boiloff vapors during propellant transfer operations.

4.5.4 Rapid OTV Response

Propellant storage can reduce OTV turnaround times and provide rapid response capability for rescue and/or other high value OTV services. It eliminates the problems associated with rapidly replanning STS manifests and flight schedules in support of some urgent mission need. Since OTV propellant is by far the largest logistics component required for any GEO mission, particularly a rescue mission where the stage hardware is already at the SOC, having immediate access to sufficient quantities of propellant is of tremendous importance and value.

4.5.5 OTV Sizing

Another area of potential improvement which is synergistically related to propellant storage on SOC is OTV sizing and technology enhancement. To fully capitalize on the benefits of propellant storage on SOC the advantages of space-basing the OTV must be considered. Together they offer new vistas of performance potential for application to SOC based operations.

Space basing the OTV means it would not have to be carried into orbit full of propellants with tank structural margins sized to boost and/or crash loads. Lighter weight tanks afforded by these factors along with lighter weight insulation if refueling in orbit is done just before mission departure (no boiloff while awaiting propellant delivery) can result in highly efficient designs with low inert weights.

These, plus continuing advances in design techniques and materials along with the possible use of lower T/Ws in recognition of the transport needs of lightweight space structures fabricated on the SOC offer as a feasible goal, stage mass fractions approaching 0.95 (propellant weight to stage gross weight). Mass fractions in this range combined with I_{sp} improvements from

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continuing engine developments can have dramatic effects on OTV sizing requirements. Further, carrying the space-based OTV stages to orbit empty would allow the design of larger stages, needing only to fit within the orbiter cargo bay length rather than be sized to the 29,500 Kg (65,000 lb) weight capacity which is much smaller.

One possibility introduced by these advances is the potential for a common propulsive stage solution to the potentially separate needs of cargo OTVs, manned OTVs, and manned rescue.

Figure 4.27 interrelates the main performance parameters involved in this central stage commonality issue. The main parameter catagories are: (1) delivery performance, (2) stage characteristics, and (3) propellant requirements. The data are keyed to the GEO rescue mission which is expressed in terms of single stage round trip (r.t.) payload capability. The graph shows the one way delivery capability (stage returns to SOC empty) of this same stage and the r.t. capability of two of these stages operated in a tandem arrangement (both stages return to SOC, one empty and the other with the r.t. payload). The bottom of the graph shows the propellant requirement for this common stage design as functions of Isp and mass fraction.

As seen on the figure the one way payload is in the 20,000 to 30,000 lb range and the r.t. payload falls in the 15,000 lb plus range, both close to the size ranges projected in various industry/government mission models for

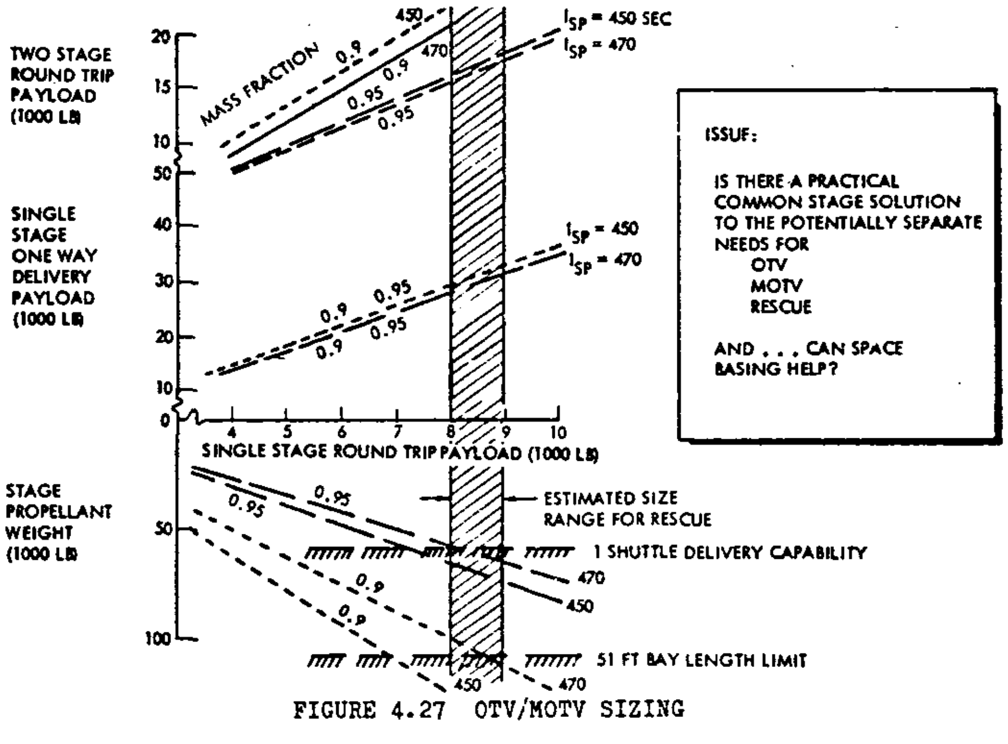

FIGURE 4.27 OTV/MOTV SIZING

these types of payloads. Of perhaps even greater significance is the propellant requirements for this advanced common stage design. The high I_{sp} and mass fraction afforded by a SOC based design permits the stage propellant requirements to be met with a single Shuttle delivery flight... a significant and desirable achievement.

However, if a high technology design must be deferred because of early funding limits, SOC basing also offers the versatility for utilizing a low technology approach. A stage configured to fit the orbiter bay length constraint, say 51 feet allowing for the docking module and end clearances, could be designed to conventional mass fractions and I_{sp}s, thereby saving development costs. This stage configuration could perform the above three types of missions.

Thus, space-basing on SOC offers the potential for major advances in OTV performance and at the same time provides the versatility to retain and perhaps enhance the viability of low technology OTV design approaches.

Achievable OTV performance improvements can produce large savings in propellant logistics costs. This is illustrated graphically in Figure 4.28. Propellant savings from 8,600 to 11,000 Kg (19,000 to 25,000 lb) depending upon I_{sp} are shown for a mass fraction improvement from 0.9 to 0.95. This represents approximately a 25% reduction in propellant delivery flights and associated logistics. However, to achieve the full benefit of these savings the OTV design must be interacted with the SOC design. Impacts of space servicing and other SOC operations on OTV design must be understood and weighed against the expected performance improvements to

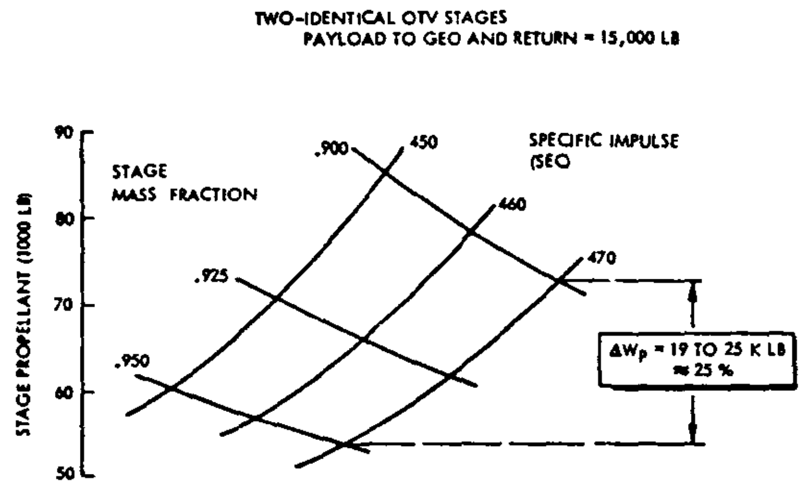

FIGURE 4.28 PROPELLANT SAVINGS WITH SPACE-BASED OTV DESIGN

produce the most effective concepts meeting the national space program needs. The potential for accomplishing these objectives appears to be emerging through SOC-OTV synergistic interactions. Although much needs to be studied and many technical issues must be solved, the payoff in future benefits strongly suggests this as a promising new arena of study.

4.6 EXTERNAL TANK PROPELLANT RECOVERY

Techniques developed in this study for recovery of LOX and LH_2 residuals from the Shuttle ET are considered feasible and show major program benefits through synergistic interactions with other SOC functions.

4.6.1 Previous Studies

Many techniques have been proposed for recovery of LOX and LH_2 residuals from the ET in order to supplement the orbiter tanker loads used to resupply on-orbit propellant depots and space-based OTVs. These approaches typically involve taking the ET into orbit for rendezvous with the depot or OTV prior to transfer of the residual ET propellant.

In the process, however, aerodynamic drag and capillary forces will cause the propellants to entirely rewet the warm walls of the ET tank with the result that all or most of the liquid would boiloff in the six hour zero-g period prior to rendezvous. Approaches for adding ET insulation or internal isolation tanks proved to be heavy and cost-ineffective (Reference 1). Also, the flight performance penalties for boosting the ET to 370 KM (200 n.mi) orbit (versus the normal 57 n.mi separation altitude) and later de-orbiting the ET were appreciable. Because of these problems, ET propellant recovery was not previously seen as attractive.

<u>Alternate Approaches</u>

An investigation was made into the possibility of recovering ET residuals during an extended sub-orbital coast prior to ET separation. Boiloff losses would be greatly reduced by a slight settling thrust applied after MECO to keep the propellants from lofting forward while they were being rapidly transferred to insulated tanks in the cargo bay. Results showed that this concept was feasible and involved little or no flight performance loss and only a minor amount of RCS propellant for settling impulse. The average weight of propellant recovered on a fully loaded (29,500 Kg) SOC resupply flight was approximately 4,300 Kg (9,500 lb). For reduced cargo loads, much more ET propellant would be available. . . almost pound for pound.

<u>Trajectory Modifications</u>

The Shuttle ascent trajectory is constrained primarily by the requirement to aim the empty ET for impact in a safe target area and at the same time satisfy abort safety requirements. As such, the orbiter with its external tank does not burn to a propellant depletion cutoff but to a guidance cutoff. For the Basic Reference Mission (BRM), which is the one closest to a fully loaded SOC resupply flight, launch is due east from KSC

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with ET separation a few seconds after MECO (main engine cutoff) at an altitude of ~ 105 KM (57 n.mi). OMS I burn starts a short while later and injects the orbiter into a 275 x 90 KM (150 x 50 n.mi) orbit. OMS II burn occurs near apogee and circularizes the orbit at 275 x 275 KM (150 x 150 n.mi). For a worst case SOC resupply, OMS I and II burns would both be extended to achieve a circular orbit of approximately 425 KM (230 n.mi) and will require full loading of the integral OMS tanks.

The Indian Ocean was chosen as the target ET impact area chiefly because of its proximity, small impact footprint and low density of ship traffic. However, because of the flat tradeoff between gravity losses and drag losses in the ascent trajectory, it appears that little or no flight performance would be lost by using a steeper ascent profile in order to target the ET for the Pacific rather than the Indian Ocean. Although the impact footprint would be larger and ship traffic might be heavier, this is not seen as a serious drawback. As shown in Figure 4.29, this would permit approximately 20 minutes sub-orbital coast of a mated orbiter/ET in a low drag environment during which propellant recovery could be accomplished. 20 minutes of RCS settling thrust at an average level of 445 N (100 lbf) would require approximately 200 Kg (450 lb) of propellant of which perhaps one-half would be effective in increasing orbiter ΔV, thereby reducing the OMS propellant required to reach final orbit. Two of the more promising system options for providing settling thrust are as follows:

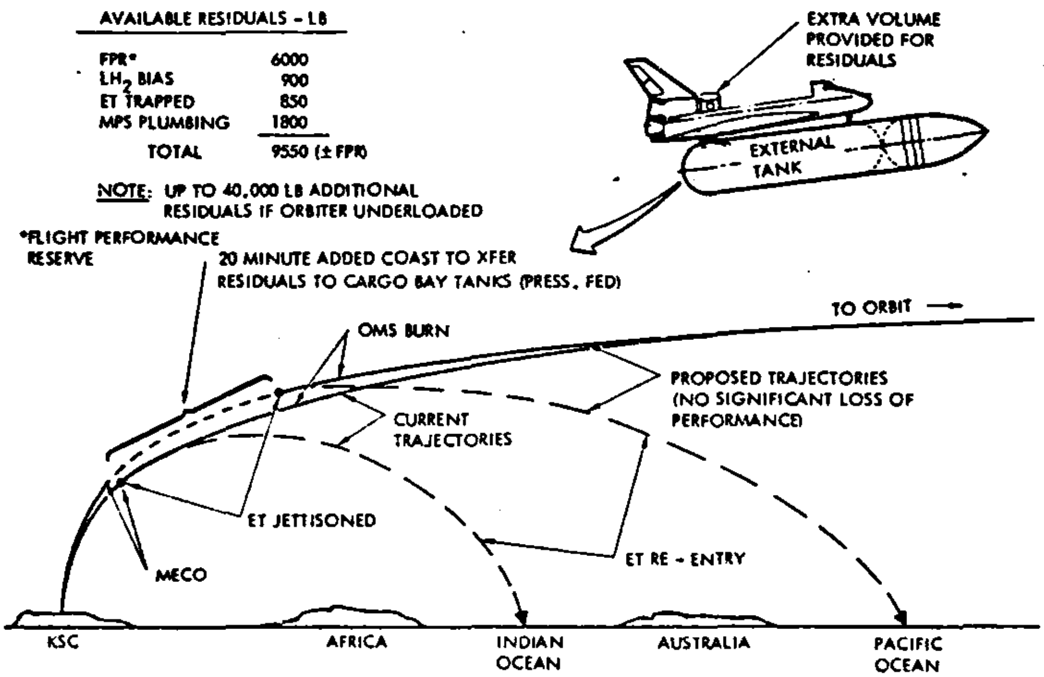

FIGURE 4.29 MODIFIED TRAJECTORY CONCEPT FOR ET PROPELLANT RECOVERY

- Addition of throttling capability to two or more of the present aft firing RCS thrusters on the orbiter

- Addition of two or more vernier RCS engines on the orbiter in an aft-firing direction.

Aerodynamic drag at MECO is approximately equal to the thrust of one RCS engine (3870 N or 870 lb) and would rapidly decline as higher altitudes are reached in the mated sub-orbital coast period. Initially, two RCS thrusters would be used to settle and stabilize the ET propellant surfaces. If throttling capability is provided, their thrust would gradually be reduced to about 220 N (50 lb) total as aerodynamic drag decreases. If six vernier RCS engines are provided in an aft firing position, the thrust would be decreased in steps from 7700 to 3900 to 630 to 440 to 220 N (1740 to 48 lb). Attitude control would be provided in the normal manner by the orbiter RCS system.

Recovery Concepts for Dedicated Tanker Flights

On a fully loaded (29,500 Kg) orbiter tanker flight, ET propellant residuals at MECO can range from 1600 Kg to 7000 Kg (4300 Kg average) depending on random variations in Shuttle flight performance and boost environments. The regular LOX and LH_2 resupply tanks mounted in the cargo bay can be oversized to leave room for the higher figure ($\sim$ 36,000 Kg total) by increasing the combined length of the 7.9 M (26 ft) tank assembly by approximately 1.5 M (5 ft). Preliminary analysis shows that this propellant can be transferred through a pair of 6.3 cm (2.5 inch) diameter lines in $\sim$ 15 minutes using only the pressure difference between the ET ullage gases and the vapor pressure ($\sim$ 16 psia) of the propellants in the cargo bay tanks.

Each 6.3 cm diameter transfer line would tap into the SSME feedline plumbing in the orbiter engine compartment, pass through the cargo bay aft bulkhead and terminate in special discharge ports located in the resupply tanks so as to mix the transferred residuals thoroughly into the propellant bulkhead. Redundant isolation valving and electrical inhibit circuitry would be installed to preclude flow into or out of the SSME feedline manifolds prior to MECO.

During at least part of the transfer, propellant entering the receiver tank would be a two-phase liquid/vapor mixture, especially at the start due to transfer line chilldown and toward the end due to temperature stratification in the ET residuals. The vapor phase, however, would be readily condensed by mixing into the bulk propellant with the aid of appropriate jets in the receiver tank. At some point, depletion of the residuals would occur and most of the flow would be ET ullage vapors which would continue to be absorbed by condensation until the heating effect in the propellant bulk causes its own vapor pressure to approach that in the ET ullage. Instead of relying solely on the ET ullage pressures for transfer of residuals, boost pumps capable of handling two-phase flow could be used to expedite the transfer and/or permit use of smaller transfer lines.

During the above transfer process, heat soaking into the SSME feedline and plumbing will cause propellant trapped there to gradually boil and expel a two-phase mixture into the main transfer stream. After ET separation, this cryopumping process in the engine plumbing will continue to force two-phase propellant flow into the cargo bay tanks. The acceleration provided during OMS I and II burns can help to transfer liquid in preference to vapor. On this basis, it may be possible to cryopump say 95% of the trapped engine propellants into the cargo bay tanks before SOC rendezvous or before the pressure limit of the receiver tank is reached. This could make overboard dumping of the normally trapped engine propellants unnecessary except for the minor amounts left after closing the transfer lines to the cargo bay. Depending on the amount remaining, it may be necessary to add a small hydrogen overboard bleed line to avoid opening the main LH_2 valves in the SSMEs to accomplish hydrogen dump and thereby overspeed the unloaded turbopumps.

Transfer Concepts for Mixed Cargo Flights

As discussed in Section 4.4.5 on SOC propellant storage, most non-tanking orbiter flights tend to be underloaded or volume limited. This permits the installation of an additional propellant supply tank(s) in the cargo bay to allow utilization of excess payload capacity, thus saving a significant proportion of Shuttle launches. In the rare case where a non-tanking flight was loaded to its weight limit (29,500 Kg) with very dense cargo such as coils of aluminum sheet for beam construction, it would still pay to include an empty propellant catch tank in the cargo bay for the sole purpose of recovering the ET residuals after MECO. Figure 4.30 shows a compact toroidal tank design which can hold approximately 11,000 Kg (24,000 lb) of LOX/LH_2 at 6:1 and still fit within the 2.7 M (9 ft) length of a standard OMS kit. This could easily handle the extreme ET residual weight of 7,000 Kg (15,500 lb) at a mixture ratio as low as 3.5 to 1. For the more typical non-tanker flight having a 14,000 Kg (30,000 lb) cargo and 15,500 Kg (35,000 lb) excess capacity, either a full sized or a somewhat smaller resupply tank assembly could be used. The minimum capacity of such an assembly should be about 25,000 Kg (55,000 lb) at 6:1 mixture ratio. If its inert weight is 1,360 Kg (3,000 lb), it would be loaded with 14,500 Kg (32,000 lb) of propellant at launch, leaving 10,400 Kg (23,000 lb) capacity to accept the extreme cases of ET residuals which would be 7,000 Kg (15,500 lb) at either 4:1 or 9:1 mixture ratio. One option would be to use two of the catch tanks of Figure 4.30 with a total capacity of 22,000 Kg (48,000 lb) at 6:1 or catch tanks of a slightly longer design, 3 M instead of 2.7 M (10 ft instead of 9 ft), giving them a combined capacity of 25,000 Kg. One full sized tank and two identical catch tanks could then efficiently handle almost every cargo mixing case and still provide spare tank capacity for recovery of the extreme ET residual weight.

In the rare case where the catch tank is launched dry because all orbiter capacity is used by high density non-propellant cargo, special procedures described in the next subsection are necessary to transfer the ET residuals. This is chiefly because there is no initial liquid bulk present in the receiver tank to condense the vapor phase of the incoming flow.

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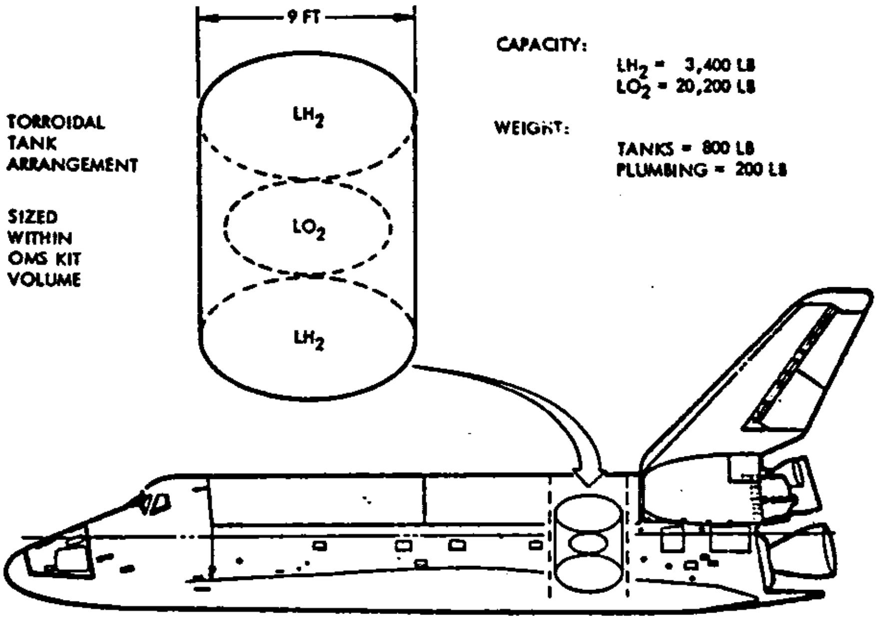

FIGURE 4.30 ET PROPELLANT CATCH TANK CONCEPT

<u>Benefits From "Dry-Launched" Supply Tanks</u>

On all tanking flights, even when fully loaded to maximum 65K lb cargo capacity, the option exists of launching with the supply tank empty and filling it after MECO with the extra propellant (almost pound for pound) left in the ET. Although this approach would impose a small penalty in the amount of propellant delivered to SOC, it can provide several major benefits in terms of abort safety, system simplification, ground operations and OTV evolution, as follows:

- o <u>Orbiter Engine-Out Capability</u> - With the propellant supply tank empty, the orbiter would be launched unloaded by about 28,000 Kg (62,000 lb) and substantially earlier engine-out times could be tolerated for RTLS and AOA abort, and also for SOC rendezvous. In the latter case the propellant load delivered would be reduced but the resupply mission would not be entirely lost. The corollary option of launching with the supply tank full and feeding the orbiter Main Propulsion System (MPS) in case of an engine-out condition is not recommended because of the system complexity, reliability and safety problems involved, and the large transfer lines required to feed the operating engines.

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- Abort Safety - If an RTLS or contingency orbit is required, the propellant supply tank would already be empty and no rapid dumping or crash landing with propellants aboard would be needed.

- System Simplification - No provisions for emergency dumping of propellant would be required with attendant large line sizes, dump port connections, an emergency helium pressurization system, wake recirdulation problems, etc. Figure 4.31 shows the complex provisions required for a similar situation of transporting a loaded Centaur stage in the orbiter.

  During ground operations no cryogenic fill/drain/vent system would be required for the supply tank, and no purge bag or helium flow would be required for the supply tank MLI insulation.

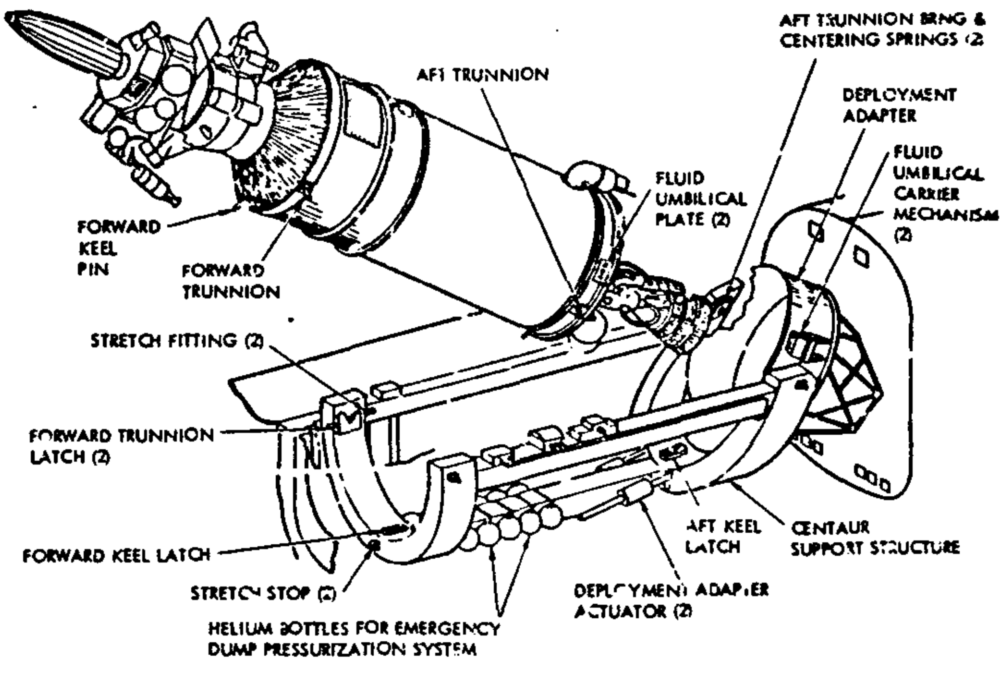

FIGURE 4.31 CENTAUR INSTALLATION FEATURES IN ORBITER BAY

- <u>Inert Weight Reduction</u> - In addition to eliminating the system weight items mentioned above, the structural weight of the supply tank and its elaborate mounting provisions in the cargo bay could be greatly reduced due to the absence of internal propellant mass during launch or crash landing conditions. A slight gas pressure of 3 psig would be used in the tanks at launch to provide pressure stiffening of the tank and prevent fluttering of its walls in the Shuttle acoustic environment.

- <u>OTV Benefits</u> - Where ground-based OTVs such as Centaur are carried in the orbiter, all of the above advantages in crew safety and system simplification would also apply. If new-generation ground-based OTVs are built, their mass fraction can be made equal to that of a space-based OTV by the above reductions in structural weight. The same OTV design could later be used for space-based OTVs, thereby providing another evolutionary option for SOC. Also, whenever space-based OTVs are transported to SOC, they can be filled after MECO and delivered in a full condition, thereby saving a later orbiter tanking flight.

Most of the advantages obtainable with dry launching also apply in cases of cargo mixing where a partial load of propellant is added to the cargo manifest to utilize the full capacity of the orbiter.

<u>Trajectory Modifications Required</u>

A dedicated tanker flight with a dry-launched supply tank can involve the transfer of up to 36,000 Kg (80,000 lb) of propellant from the ET after MECO, compared to 7,000 Kg (15,500 lb) maximum for a "wet" launched case. The most promising option for transferring this increased amount of propellant is to increase the mated sub-orbital coast period from 20 minutes to approximately 50 minutes by targeting the ET for a "once-around" impact in the Atlantic Ocean rather than the Pacific Ocean. Transfer flow rates would also be increased by enlarging the transfer line size from 6.3 cm (2.5 inch) diameter to $\sim$ 8.9 cm ($\sim$ 3.5 inch) diameter.

Once-around impact in the Atlantic could be accomplished by modifying the Shuttle ascent trajectory to inject the orbiter/ET combination into a low elliptical orbit at MECO with its perigee such that ET re-entry and impact would occur in the target area. The additional ΔV required would come out of the 28,000 Kg (62,000 lb) or so of useable residuals left in the ET from launching the orbiter in an unloaded condition. Part of the additional ΔV required would come from the RCS settling thrust needed to position the ET propellants over the tank outlets during the 50 minute transfer period. As previously described, the RCS system modifications required for this task are fairly minor. If an average RCS thrust of 445 N (100 lbf) is provided for 50 minutes, the propellant consumed would be approximately 500 Kg (1100 lb), providing an additional 4.5 Mps (15 fps) of orbiter ΔV, assuming that one-half of the impulse is lost to orbital drag and acceleration of the ET inert weight. No problem is seen with the present total capacity of the aft RCS tanks and OMS integral tanks (interconnected) since the main Shuttle engines will provide a higher initial orbit at MECO than for a normal SOC rendezvous mission.

Although the present OMS and RCS systems are capable of taking an empty ET to orbit, attitude control with 29,000 - 35,000 Kg of propellant inside the ET may require modification of the RCS software and/or realignment of the aft firing RCS engine thrust vectors to accommodate the new c.g. conditions. Prior to releasing the ET for re-entry, a small amount of RCS or OMS impulse may be desireable for final aiming of the ET into the impact target area. This would be followed by an orbiter pull-up maneuver during OMS I burn which establishes the final orbit apogee and then an OMS II burn for circularization. The overall flight performance penalty of the above flight sequence is believed to be relatively minor, but detailed analysis is required to confirm this.

Special Transfer Procedures

Propellant transfer into a dry tank is essentially the same as described earlier for a "wet" tank, except that the chilldown boiloff vapors cannot be absorbed and must be vented overboard. High velocity swirl jets direct the incoming two-phase flow tangentially so as to convert the tank into a combination centrifugal liquid/vapor separator and flash-chamber subcooler. When the tank is chilled and an initial mass of subcooled liquid is present to act as a thermal sink, venting is stopped and the incoming two-phase flow is absorbed by mixing and condensation with the subcooled contents. Transfer can continue until the ingestion and condensation of ET ullage vapor causes enough heating of the propellant bulk to exert a vapor pressure which equals the ET ullage pressure. If thermal analysis shows that a high recovery of liquid ET propellant is not possible, a boost pump capable of handling two-phase flow could be activated to supplement the transfer up to the pressure limit of the receiver tank.

Overall Evaluation of Dry-Launch Method

Other factors being equal, dry launch of a dedicated full-sized propellant supply tank would reduce the propellant delivered to SOC by about 1,700 Kg (3,700 lb), since the conversion factor of orbiter weight to ET residual is approximately 0.94. Most of this penalty, however, could be offset by reductions in the inert weight of the orbiter tank and its supporting cradles, and elimination of the emergency pressurization and dump system described earlier. Another drawback for dry-launching is the inability to deliver deeply subcooled propellants. These can be subcooled, however, after delivery if active refrigeration is provided on SOC.

In spite of such drawbacks, it appears that the advantages of abort safety, system simplification and OTV performance improvements listed earlier could weigh decisively in favor of using the dry launch approach for propellant resupply and OTV transport.

Technology Assessment of ET Propellant Recovery

The development risks involved in the various techniques described in this report for transferring propellant from the ET, are considered to be relatively low. A moderate amount of ground testing and some subscale low-g or zero-g testing will be required to verify performance of the swirl jet and mixing nozzle system design. Such testing should be considered for inclusion in any future in-orbit fluid transfer testing such as the LeRC 100 lb LH_2 transfer experiment planned as an orbiter payload.

4.7 OVERALL LOGISTIC SAVINGS

Preceding sections discussed three major areas of space operations logistic savings made possible by SOC.

- o Propellant Storage
- o Recovery of ET Propellant Residuals
- o Space-Based Versus Ground-Based OTVs

Because of synergistic interactions with SOC these factors have the potential of saving up to one-half of the Shuttle flights otherwise required for SOC operations.

Figure 4.32 illustrates a moderate application of these factors and shows the overall savings which would result for a sample SOC traffic model in the 1990-1995 time frame. A total of 14 Shuttle launches (not counting OTV mission round-off flights) are shown representing typical yearly STS requirements for normal SOC supply operations.

Four resupply launches are assumed at 90-day intervals for transporting four logistic modules weighing approximately 9,000 Kg (20,000 lb) each. These flights are neither weight limited nor volume limited, and recovery of 11,000 Kg (24,000 lb) of ET propellant residuals per flight is assumed through use of a single catch tank in the cargo bay as shown in Figure 4.30. This assumption is conservative since there would probably be room for two catch tanks or even a full-sized propellant tank assembly (28,000 Kg capacity) which could be pre-loaded with 19,000 Kg of propellant makeup, a total 29,500 Kg cargo and recover an additional 4,000 Kg of ET residuals.

Three launches of low density construction materials are assumed per year, but no ET propellant recovery is credited because the cargo is volume limited. This also is conservative because it would be possible to split that cargo up among four or more launches, leaving room for propellant tankage to bring all the cargo loads to a full 29,500 Kg and still recover an additional 4,000 Kg of ET residuals per flight.

One MOTV mission is assumed per year supported by four Shuttle launches loaded to 29,500 Kg, containing three and a half tanker loads and a manned OTV payload capsule. With such high density cargo, room is available for a

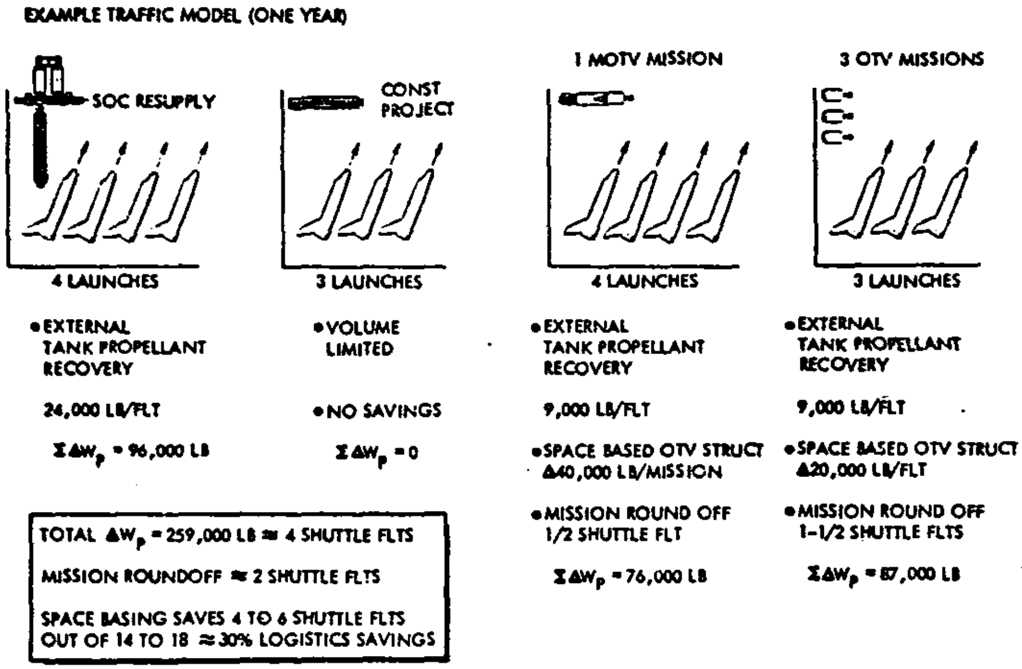

FIGURE 4.32 POTENTIAL LOGISTICS SAVINGS - SOC BASED OTV

"catch tank" to recover 4,000 Kg average of ET propellants per launch. Credit is taken for 18,000 Kg of MOTV propellant requirement saved by use of space-based MOTVs. This is based on an MOTV mass fraction approaching 0.95 versus 0.90 for the typical ground-based design which must withstand the harsh structural dynamic loads of launching with propellants in its tanks.

Three unmanned OTV missions were assumed in Figure 4.32, each supported by a full tanker load of propellants and recovering an additional 4,000 Kg of ET residuals. A 9,000 Kg saving in the propellant required for each OTV mission was taken, based on a space-based OTV mass fraction of 0.95 versus 0.90 for ground-based designs.

The total propellant savings shown in Figure 4.32 is 117,000 Kg or approximately four Shuttle flights' worth. In addition, one-half of a mission round-off flight was assumed saved for each OTV and MOTV mission, resulting in additional savings of two flights per year. Based on this application of techniques, it is seen that four to six Shuttle launches out of 14 to 18 (approximately 30%) can be saved in the traffic model of Figure 4.32. Using all of the available techniques discussed earlier, overall savings could approach 50%. No serious technical difficulties in achieving such results are foreseen, and further study is recommended.

4.8 RESUPPLY OF NON-CRYOGENIC FLUIDS

General techniques were studied for resupply of SOC non-cryogenic fluids such as hydrazine, water, fluorocarbons and helium. This analysis took into account the factors of:

- o Usage Rate
- o Vapor Pressure at Storage Temperature
- o Toxicity and Contamination Characteristics
- o Safety Issuos

In general, fluid transfer through disconnect fittings into permanent SOC tankage is preferred for large resupply requirements such as hydrazine, while container replacement is recommended for smaller amounts. In either case, diaphragmed tanks operating in the blowdown mode appear to provide the simplest and most practical means of storage, guaging and distribution for SOC fluid systems. Typical spacecraft hydrazine systems use a 3:1 blowdown pressure ratio in which the tankage is initially filled to 2/3 capacity with the remaining 1/3 volume pressurized with GN2 to $\sim$ 2.4 x 10^6 N/M^2 (350 psi). Capillary-type liquid acquisition devices can be used instead of expulsion diaphragms; however, close monitoring of the tank liquid quantity is necessary to prevent entry of pressurant gas into the downstream system at liquid depletion. Simple PVT (pressure, volume, temperature) tank guaging methods should be adequate in any case.

Where replacement of tanks is used for fluid resupply, manifolding of two or more tanks in the SOC fluid system (along with isolation valves) is recommended for redundancy. Plumbing, valves, tanks and disconnects should be installed externally to the habitability modules and other enclosed areas to mirimize hazards for leakage and spillage. Use of the RCM with EVA operation of disconnect fittings, is recommended for tank replacement. The empty containers can be used for fluid waste disposal and/or returned to earth. Another option is to reprocess them in orbit as raw construction materials, or connect them together to form structural elements such as strongbacks, columns, etc.

In the special case of resupplying high vapor pressure liquids such as fluorocarbons or refrigerants, no pressurizing source is usually required other than the vapor pressure of the fluid itself. Also, if the fluid is not needed immediately in the vapor-free liquid state, no expulsion diaphragms or zero-g capillary devices are necessary.

No special procedures are required for resupply of gases other than normal safety precautions. Transfer can be accomplished by a simple tank-to-tank blowdown procedure, or can be aided by a SOC-mounted gas compressor to save weight in the orbiter supply tank(s).

4.8.1 Hydrazine Resupply System Design

Hydrazine resupply requirements for SOC are estimated to be on the order of 2,300 Kg (5,000 lb) for 90 days of usage, which includes SOC attitude control, orbit makeup and OTV requirements at the flight support facility. The task of resupplying hydrazine in-orbit has been analyzed in depth by Rockwell under a GSFC study for refueling MMS spacecraft, as reported in Reference 5. The design resupply requirement for this study was also for 2,300 Kg of hydrazine, and the basic system analysis and concepts are considered to apply closely to SOC.

Safety Considerations

The basic safety requirements for hydrazine resupply are covered in NASA Handbook 1700.7 (Reference 4). In addition to the general safety policies outlined for cryogenic propellants in Subsection 4.4.4 of this report, special design and operating procedures will be provided to insure that propellant detonation can not occur due to rapid adiabatic compression of hydrazine vapors.

Basic Transfer Methods

The principal means available for transferring hydrazine propellant in-orbit are:

1. Pressurized Transfer
   - Single Stage Blowdown
   - Multi-Stage Blowdown
   - Regulated Pressure Expulsion
2. Pumped Transfer
   - Ullage Compression
   - Ullage Displacement

These concepts are shown schematically in Figure 4.33 for refueling a blowdown type spacecraft (S/C) receiver tank having a maximum working pressure of 2.4×10^6 N/M^2 (350 psi). The single stage blowdown concept (a) is the simplest but requires the highest working pressure for the hydrazine supply tank of the ORS (Orbital Refueling System). The multi-stage blowdown system (b) is better in this respect but requires considerable redundant valving and a complex procedure for safe operation. The regulated pressure system (c) and pumped transfer system (d) are both moderate-weight, straightforward concepts. The ullage displacement pumped transfer concept (e) is often proposed for zero-g application but is complex and has no inherent advantages for refueling a blowdown type receiver tank.

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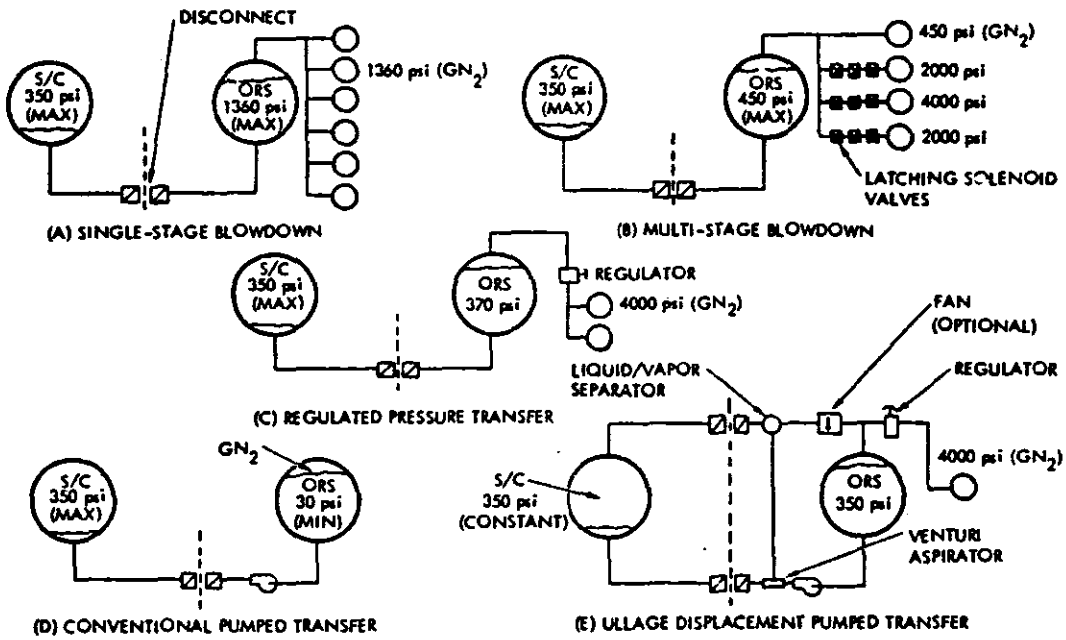

FIGURE 4.33 CANDIDATE PROPELLANT TRANSFER METHODS

It was therefore eliminated from further consideration. An ROM system weight and cost comparison was performed for the remaining four concepts as shown in Table 4.2. This included separate evaluations for each of the candidate supply tank designs shown in Figure 4.34. Although the pumped transfer concept gave the lightest system, poor pump development status and electrical power requirements were seen as significant drawbacks. Considering all factors, the regulated pressurized transfer concept was selected as having the best combination of weight, cost, reliability, safety, development status and operational simplicity.

Tankage Selection

Figure 4.35 excerpted from Reference 5 shows a regulated pressure transfer refueling system installed in one of three orbiter cradles planned for servicing and/or transporting a MK II MMS spacecraft. The refueling disconnect is mounted on the docking ring in the center cradle. The five TDRSS hydrazine supply tanks (450 Kg or 1,000 lb capacity each) have diaphragms, which greatly simplifies ground and orbital transfer operation. An alternate tankage design is shown in Figure 4.36 using four VO-75 (Viking Orbiter) tanks (680 Kg or 1,500 lb capacity each). Although this tank is not presently equipped with a diaphragm, the supplier (PSI) considers that tank to be state-of-the-art and has offered to develop one on a nominal

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TABLE 4.2 ORBITAL REFUELING SYSTEM TANKAGE AND TRANSFER MODE COMPARISON

SUPPLY TANK DESCRIPTION	NO. OF SUPPLY TANKS	% FILL AT 5000 LB [illegible]	TRANSFER MODE*	MEOP PRESSURE (PSI)	MEOP PRESS. W/ SAFETY (PSI)	PROPELLANT MANAGEMENT	DRAIN PORT LOCATION	AUXILIARY PRESSURE TANKS (NO. & SIZE)	WT. TANKS (LB) (1.5 BURST)	WT. TANKS TOTAL (LB)	[illegible] (LB)	WT. VALVES, REGULATORS, PUMPS (LB)	TOTAL WT. (LB) UNIQUE ITEMS	COST, TANKS (K$)	COST, TANKS TOTAL (K$)	COST, BLOWDN, REG'S OR PUMPS (K$)	TOTAL COST (K$) UNIQUE ITEMS
TDRSS 31.8×40.2 IN. OBLATE ELLIPSOID	5	50	SBD	1360	450 AT 1.5 BURST	DIA-PHRAGM	DOWN	6-25.2 IN. 4 KPSI	5 - 250, 6 - 78	1654	190	--	1844	5@127, 6@15	725	--	725
			MBD	450				6-25.2 IN. 4 KPSI	5 - 76, 6 - 78	852	175	27	794	5@130, 6@15	560	9 @ 15	698
			REG.	370				2-25.2 IN. 4 KPSI	5 - 76, 2 - 78	536	160	15	711	5@100, 2@15	530	3@12.5	566
			PUMP	450				2-25.2 IN. 4 KPSI	5 - 76, 2 - 78	536	15	30	581	5@100, 2@15	530	3@150	830
V075 55.6×36 IN. CYLINDER	4	70	SBD	1040	575 AT 1.5 BURST	SCREEN CHANNEL	AFT	4-25.2 IN. 4 KPSI	4 - 160, 4 - 78	986	210	--	1196	4@250, 4@15	1060	--	1060
			MBD	575				3-25.2 IN. 4 KPSI	4 - 90, 3 - 78	594	205	18	817	4@200, 3@15	845	6@15	935
			REG.	370				2-25.2 IN. 4 KPSI	4 - 90, 2 - 78	516	190	15	721	4@200, 3@15	830	3@12.5	855
			PUMP	300				--	4 - 90	360	20	30	410	4@200	800	2@150	1100
OMS 94×49 IN. CYLINDER	1	84	MBD	450	313 AT 1.5 BURST	SCREEN BLANKET & CHANNELS	SIDE	3-25.2 IN. 4 KPSI	1 - 495, 3 - 78	667	190	18	877	1@450, 3@15	495	6@15	705
			REG.	370				2-25.2 IN. 4 KPSI	1 - 355, 2 - 78	511	182	15	708	1@450, 2@15	480	2@12.5	705
			PUMP	300				--	1 - 302	302	19	30	351	1@450	650	2@150	950
LMSC 76-IN. DIAMETER SPHERE	1	60	MBD	400	400 AT 1.5 BURST	SCREEN CHANNELS	AFT	3-25.2 IN. 4 KPSI	1 - 625, 3 - 78	859	231	18	1110	1@435, 3@15	480	6@15	570
			REG.	370				2-25.2 IN. 4 KPSI	1 - 625, 2 - 78	781	225	15	1021	1@435, 2@15	480	2@12.5	505
			PUMP	300				-	1 - 625	625	25	30	680	1@435	480	2@150	780

*IN TRANSFER MODE COLUMN, SBD = SINGLE BLOWDOWN
MBD = MULTI-BLOWDOWN
REG = REGULATED

NOTE: INCLUDES AMORTIZED RETOOLING COSTS

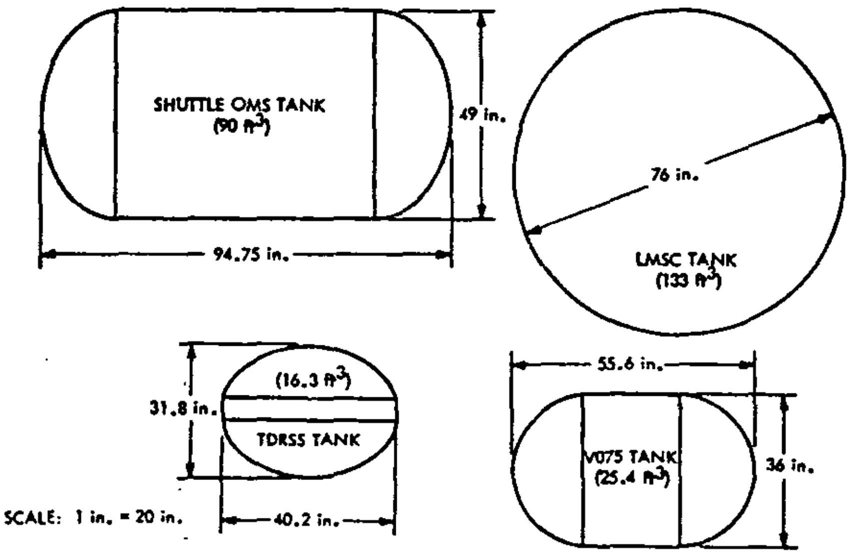

FIGURE 4.34 CANDIDATE TANK GEOMETRY COMPARISON

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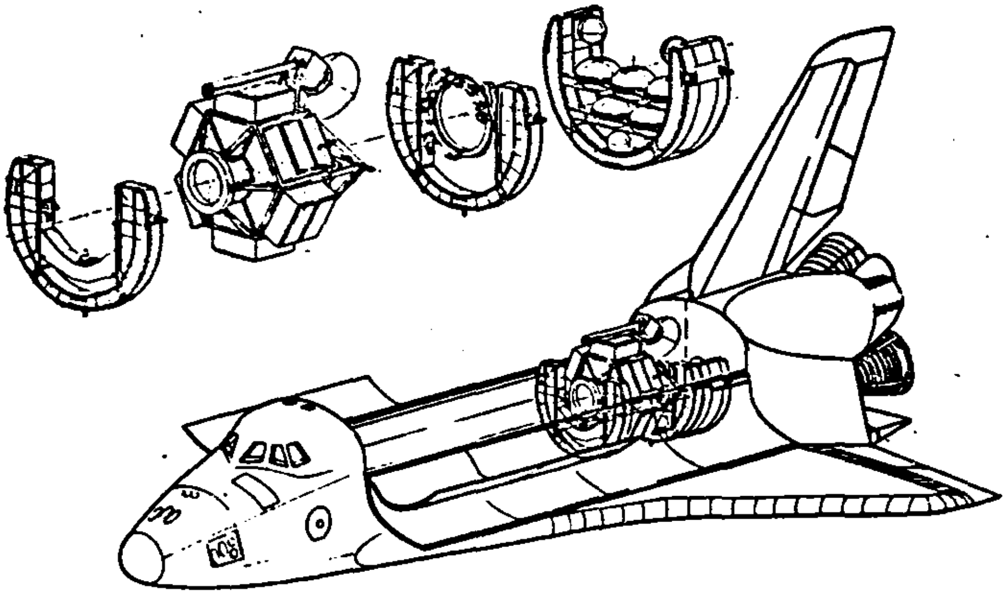
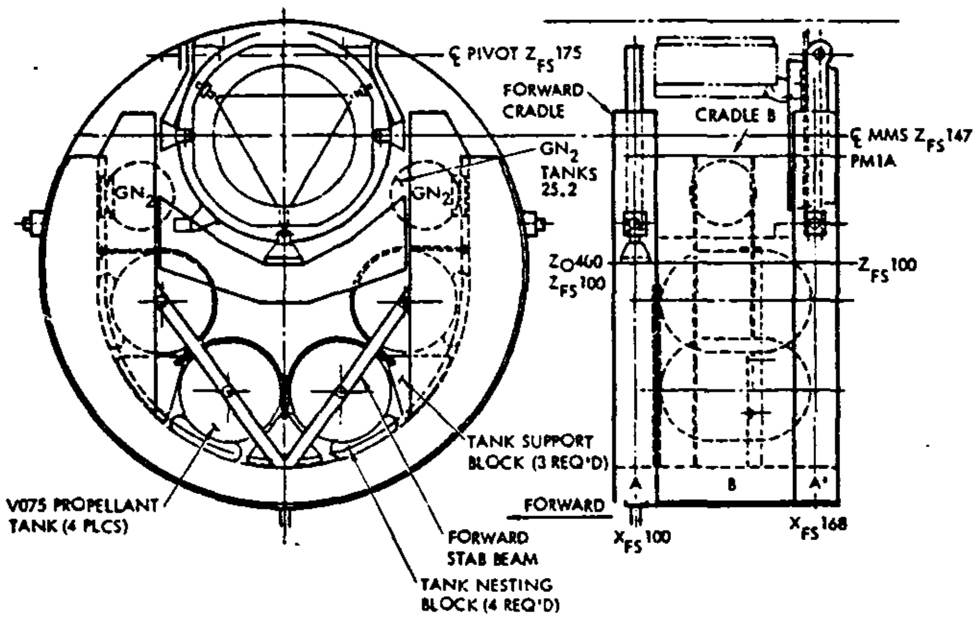


FIGURE 4.35 MARK II AND FSS INSTALLED IN ORBITER



MMS MARK II AND PM-1A MAY BE SERVICED WITH SAME INSTALLATION (AFT CRADLE REPLACES FORWARD CRADLE) (TANKS IN CRADLE B)

FIGURE 4.36 V075 CLUSTER PM-1A/MARK II COMPATIBLE fixed-cost basis. Room is available within the orbiter 4.6 M (15 ft) diameter envelope to add four more VO-75 tanks and associated GN2 bottles within the same cradle length. This would allow carrying a 180 day supply in one trip if that is desired. In any case, the SOC storage tankage would be sized for 180 days capacity and could use the same diaphragmed tanks as the resupply system. An alternative way to the use of a dedicated orbiter cradle would be to mount the resupply tanks on the SOC logistics module itself. Although use of a single Shuttle OMS tank or an LMSC tank is possible, they are not recommended because they are awkward to install in the cargo bay and diaphragms of that size are not state-of-the-art. Capillary devices would have decided disadvantages in this application since multiple outlets must be provided for draining in both the vertical (launch) and horizontal (landed) positions.

Instead of an emergency overboard dump system for abort situations, which may involve the considerable expense of a dedicated dump port on the orbiter, it is recommended that the tank installation be designed to withstand emergency landing loads with propellants contained. Although a peak cargo bay temperature of 200°F can occur after abort landing at a contingency field, this will not overpressurize the tanks, since they will be launched unpressurized and the vapor pressure of hydrazine is relatively low.

Refueling Disconnect Design

An industry survey was made during the preparation of Reference 5 to identify existing disconnect hardware suitable for hydrazine refueling in orbit. The best design found was the LEM (Lunar Excursion Module) glycol disconnect shown in Figure 4.37, which can be adapted to SOC hydrazine requirements with minor modification. This would consist chiefly of changing the elastomer seal material to hydrazine compatible EPR (ethylene propylene rubber). The disconnect line size (0.95 cm or 3/8 inch o.d.) and flow capacity are adequate to transfer 2,300 Kg (5,000 lb) of hydrazine in two hours, using a regulated pressure transfer at 2.5×10^6 N/M^2 (370 psi).

The disconnect of Figure 4.37 has self-sealing poppets in both halves which are opened when the halves are forced together. Angular misalignments can be tolerated and there is no integral latching mechanism which could malfunction and thereby prevent disengagement. The disconnect halves can be mounted in retracted position on the orbiter/SOC docking interface and engaged remotely by a separate actuator such as the existing MMS umbilical drive mechanism shown in Figure 4.38. Its reliability features include an over center locking mechanism and a redundant electrical motor drive. In the event of a malfunction the drive base can be demounted with standard EVA tools to effect safe disengagement of the disconnect halves. Redundant shutoff valves are provided on both sides of the disconnect for leak protection.

Hydrazine Transfer Procedures

Prior to orbiter/SOC docking, the redundant isolation shutoff valves

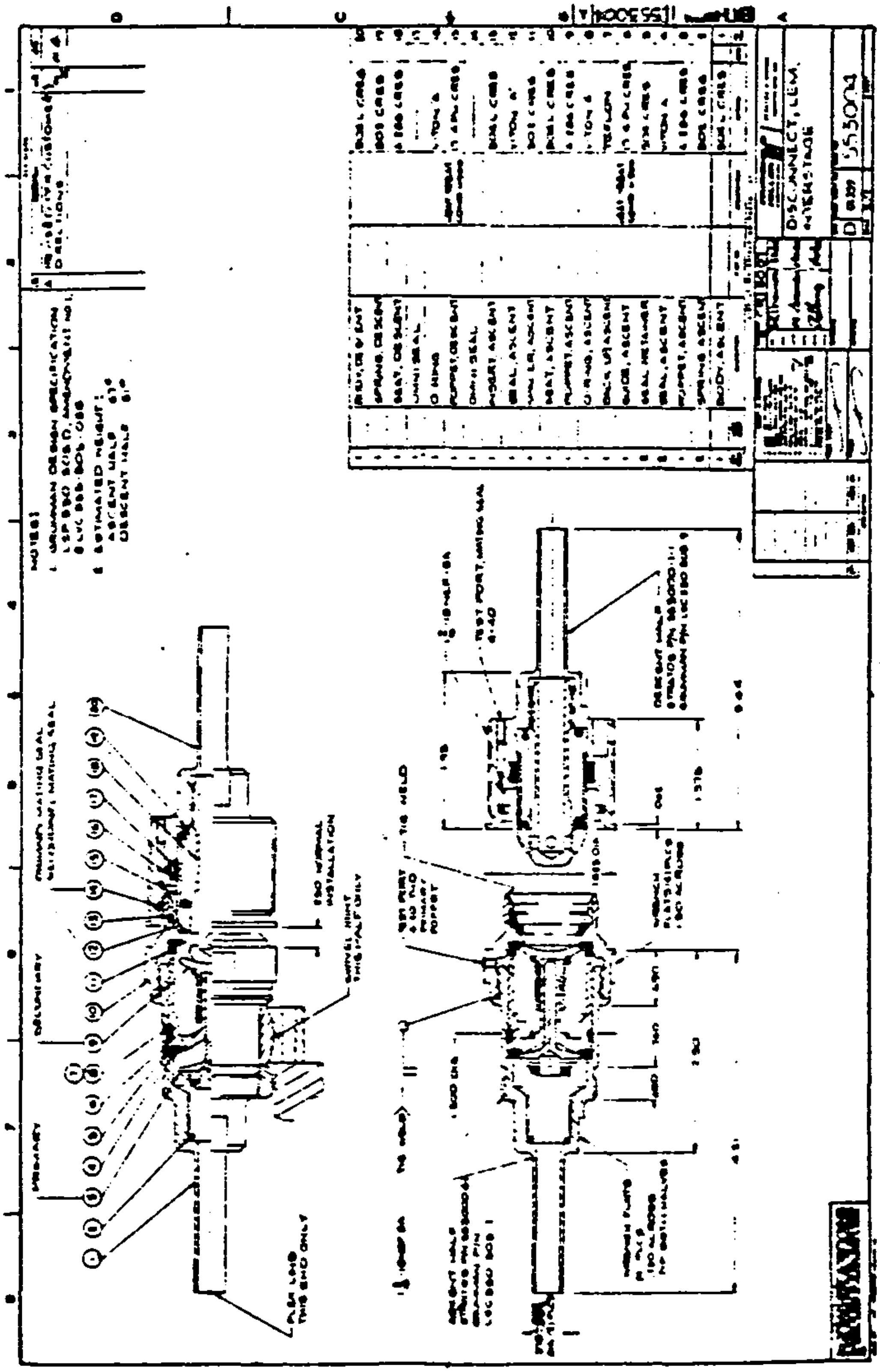


FIGURE 4.37 DISCONNECT, LEM, INTERSTAGE

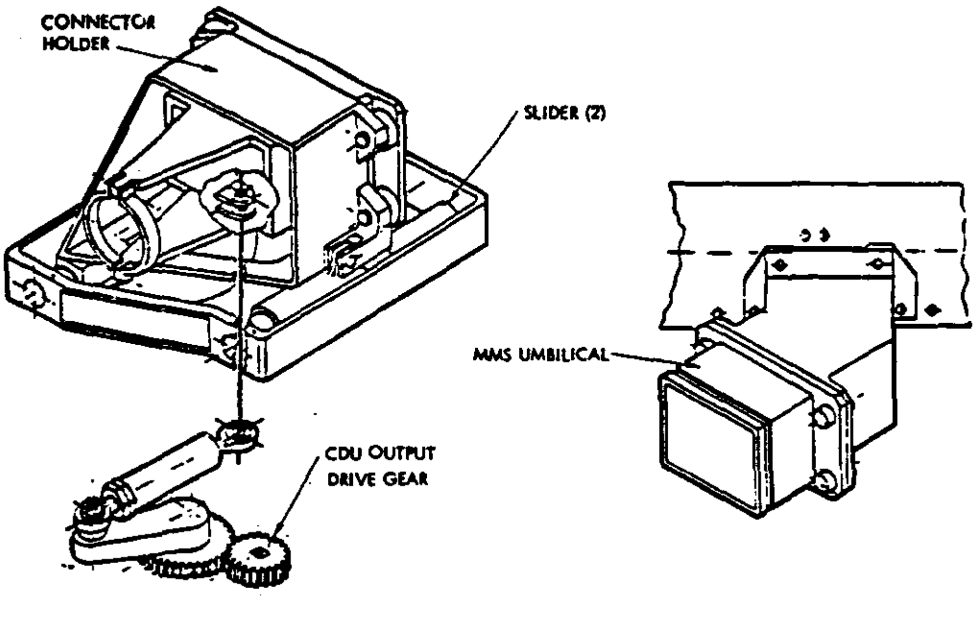


FIGURE 4.38 UMBILICAL ACTUATOR DRIVE MECHANISM

next to each disconnect half are in the closed position and both halves of the disconnect are evacuated. After docking, the retracted half on the SOC side is moved by the remote drive mechanism to fully engage the two evacuated halves. The outer interface seal is then leak tested with GN_2 through a test port provided on the SOC half. After this test, one of the isolation valves in the main flow line is opened admitting a small slug of hydrazine into the disconnect, and any leakage through the inner interface seal is stopped by the outer seal and detected at the gas test port. After this test is passed, the supply tanks are pressurized to approximately 2.5×10^6 N/M^2 (370 psi) and the remaining shutoff valves near the disconnect halves are opened, allowing hydrazine to transfer into the SOC receiver tanks whose pressure can be as low as 6.9×10^5 N/M^2 (100 psi) after blowdown to an empty condition (versus 2.4×10^6 N/M^2 when fully loaded). As transfer takes place, the ullage trapped above the diaphragms of the SOC receiver tanks is compressed and the heat of compression must be dissipated into the tank and liquid bulk before complete transfer can take place at the supply pressure of 2.5×10^6 N/M^2. The completion of transfer can be verified by use of integrating-type flowmeters and/or receiver tank pressure/temperature data. Isolation valves should be provided for each of the individual receiver tanks and supply tanks to permit isolation of any leaks and permit individual maintenance or tank replacement. Also, this provides the flexibility of loading or depleting the tanks independently as well as together. In any case, no damage will result from full system pressure across the diaphragms in either direction.

After completion of transfer, the disconnect isolation valves are closed and the disconnect halves evacuated by opening a bleed line leading to a remote balanced-thrust venting nozzle at a safe location on SOC. The SOC disconnect half is then retracted and the orbiter supply system secured. Because hydrazine freezes at approximately 35°F, either heaters or passive thermal control must be provided for all lines and tankage.

4.9 ZERO-G GAGING

A survey was made of the principal zero-g propellant guaging techniques that could be utilized for SOC. There are listed below in approximate order of development status:

- PVT (pressure, volume temperature)
- Nuclear Gaging
- RF (radio frequency)
- Acoustic Resonance
- Ullage Compliance
- Capacitance (electrical) Gaging
- Fiber-Optics

The use of settling thrust to facilitate fluid guaging was ruled out because of potential interference with other SOC operations. For any fluid exerting a low vapor pressure at its typical storage temperature (hydrazine, water, etc.), the most practical and proven guaging method is PVT, regardless of whether diaphragms or capillary devices are used for fluid positioning. In the full-tank condition, system accuracy can be as good as 0.5% with sophisticated instrumentation and data processing techniques. Where high vapor pressures are encountered (as with cryogenic fluids) the PVT method, as normally defined, cannct be used because of large errors introduced by mass-transfer (condensation and evaporation) into and out ofthe ullage space. Of the remaining methods, nuclear (gamma-ray absorption) guaging is the only one developed to the prototype stage. This method, however, is electronically complex, has been plagued by stubborn reliability problems and requires special safety procedures for handling the radioactive sources which are installed on the tank walls. The RF method uses a radio frequency antenna mounted inside the tank, along with associated electronic equipment to determine the number of RF modes or resonant frequencies in the tank as an index of fluid mass. Overall system accuracy appears to be 2-3%.

The acoustic resonance method (shown in Figure 4.39) uses an electromechanical transponder or speaker/microphone in a frequency sweep to determine the fundamental resonant acoustic frequency of the vapor/liquid/tank combination. The dominant factor in determining this

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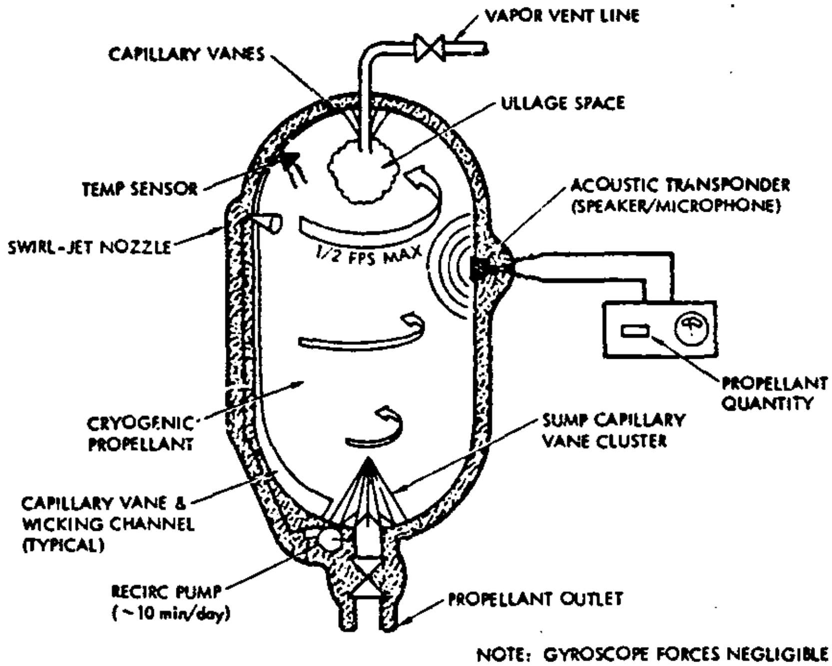

FIGURE 4.39 ZERO-G CRYOGENIC TANK GAUGING AND PROPELLANT POSITIONING SYSTEM

resonant frequency is the vapor volume, from which the liquid mass can be inferred if the tank volume is known. Propellant guaging accuracy is best in the full-tank condition and can be as low as 0.3% if all the vapor is located in a single ullage bubble in a specific position, and pressure and temperature of the ullage vapor are also known (assumed equal to the liquid). These requirements can be provided by the type of capillary propellant acquisition system shown in Figure 4.39, where capillary vanes and open wicking channels (not screen channels) are used to locate propellant at the liquid outlet and thereby displace the vapor bubbles to the vapor outlet or vent port at the opposite end of the tank. More positive positioning of the ullage space can also be provided by inducing a slight swirling action in the liquid (stronger at the vent port end of the tank) which introduces centrifugal forces, buoyancy, and axial circulation patterns in the liquid to carry stray bubbles to the ullage space at the vent outlet. The power requirements of the swirl-jet recirculation pump and its mechanical heat input to the liquid are negligible. Normally, the swirl action is required only during tank filling and later at infrequent intervals when it is desired to verify the propellant quantity, induce bulk mixing to break up temperature stratification, accurately sense bulk temperature in the tank, or prevent abnormal pressure buildup from bubbles generated at the wall by heat leakage through the tank thermal insulation.

Although the ullage compliance method can use equipment similar to the acoustic resonance method, its operating principal is not resonant frequency detection but measurement of tank pressure oscillations caused by known perturbations of tank volume. As with the acoustic resonance method, accuracy is best for the full-tank condition and can be as low as 0.3% if tank pressure and temperature are known and the vapor phase is contained in a single bubble. Tank volume perturbations for the ullage compliance method can be provided by several simple techniques:

- o Piston or diaphragm displacement
- o Venting liquid-free vapor into a small evacuated volume
- o Injection of known quantities of liquid or inert gas into the tank

It is important that the vapor phase not be dispersed in small bubbles and that the volume perturbations be accomplished quickly (<0.5 sec), to minimize errors resulting from evaporation or condensation at the ullage surface.

Gaging by electrical capacitance methods in a zero-g tank requires that the electrodes be extremely large (approximately 1/3 of tank wall area), which introduces a weight and mounting problem even if the electrode panels are bonded to the inside tank wall. Capacitance guaging systems are notoriously tempermental and sensitive to electronic drift and to minor disturbances such as low level EMI, dents, kinks, moisture and other factors which induce changes in the dielectric of its coaxial cabling and connectors.

Fiber-optic discrete-level sensors have been developed for use in positive-g environments, and have the safety advantage of not introducing electrical power into the tank. Their operating principle relies on an increase in the refractive index outside the fiber when immersed in liquid, which permits increased leakage of internally transmitted light rays. These sensors are undependable in zero-g, however, because droplets can cling to the small fiber-optic sensing surface.

In summary, it is recommended that fluids having a low vapor pressure such as hydrazine or water be guaged by the PVT method. For high vapor pressure fluids such as cryogenics, the development of acoustic resonance techniques, in conjunction with presently planned zero-g cryogenic fluid transfer experiments, is recommended. Logical choices for fall-back techniques would be the ullage compliance and RF methods. Integrating-type flowmeters can provide a useful crosscheck for any guaging system by keeping a running inventory of fluid entering or leaving a given tank.

4.10 FLUID RESUPPLY - SHUTTLE INTERACTIONS

Resupply of fluids for SOC impacts many areas of STS design and operations, e.g., ground facilities, orbiter design, logistics, prelaunch operations, boost, abort safety, orbital coast, docking and landing. Some of the principle considerations are discussed as follows:

Safety Reliability Considerations

As discussed in Sections 4.4.4 and 4.8.1, the transport and transfer of LOX, LH_2 and hydrazine propellants introduce a number of potential hazards in the areas of crew safety and mission reliability and require special design provisions and operating procedures. The controlling document for Shuttle payload safety requirements is NASA Handbook 1700.7 (Reference 4), which sets forth specific requirements and general guidelines. Some of the key areas are:

- Emergency Landing - Propellant tankage installed in the orbiter shall be capable of landing safely with propellant contained, under the 4.5g design load factors specified for emergency or abort landing in NHB 1700.7; or means shall be provided for safely dumping the propellants overboard prior to such landing. The propellant system and tankage should be able to safely withstand the soak-back temperature peak experienced in the cargo bay after landing at a contingency field not equipped with a mobile air purging unit.

Because of the large quantities of LOX and LH_2 that may be carried in the orbiter resupply tanks and the dangers that would exist in the event of a crash landing situation, unofficial plans are to provide the capability for both dumping and landing-contained, in case sufficient time for full dumping is not available as in a contingency abort situation (all SSME engines out at lift-off). Emergency dumping capability involves the addition of a high-flow pressurization system in the orbiter with sufficient helium tankage to permit dumping of 29,500 Kg (65,000 lb) of LOX/LH_2 within 6 minutes for the case of an RTLS abort.

- Redundancy - No single mechanical or electrical failure will endanger personnel or equipment. As a design goal, no single mechanical or electrical failure (except loss of power) will preclude the transfer of propellant.

- Leakage - As a design goal, three independent mechanical inhibits will be provided to prevent leakage or unplanned discharge of propellant into the orbiter cargo bay or other enclosed areas. Quick-disconnect fittings should include means for verification of interface seal integrity by inert gas leakage checks before exposure to propellants. Means should be provided for detecting external leakage from the disconnect during use, as well as from other system components. Purging capability should be provided for such contingencies.

- Monitoring - Critical system measurements should be displayed and audiable warnings and lights provided to indicate out-of-tolerance conditions. Critical valves shall have position indicators, and automated override circuitry shall be provided to automatically correct hazardous conditions through the proper sequencing of electrical commands.

- <u>Propellant Detonation</u> - Design and operating procedures shall be such that propellant detonation can not occur due to rapid abiabatic compression of hydrazine vapors.

<u>Ground Facility Modifications</u>

Provisions must be added to permit handling, loading, draining, purging, and venting of LOX, LH_2 and hydrazine propellants transported in the orbiter. Continuous helium flow will be required for insulation purging and, in case subcooled LOX or LH_2 are required, helium injection for the loaded propellant tanks in the cargo bay. A helium recovery/recycle capability may also be desired. Special fixtures for rapid installation and removal of fluid resupply tanks and associated plumbing from the cargo bay may be required for maximizing utilization of orbiter cargo capacity.

<u>Orbiter Modifications</u>

Installation of propellant and other resupply tanks will be required in the orbiter, along with the mounting cradles, plumbing, emergency dumping capability, quick disconnects and monitoring/control systems necessary to safely transport and transfer such fluids to the SOC. The LOX and LH_2 tanks will require special cryogenic insulation and helium purging, and special internal baffling in the tanks may be necessary to control slashing forces during boost.

If sub-orbital recovery of propellant residuals from the ET is to be accomplished, a specially designed catch tank and transfer plumbing system will be required, along with modifications of the RCS system to provide attitude control in a mated condition and suitable thrust for settling the ET residuals.

<u>Zero-G Propellant Acquisition and Guaging</u>

Moderate development effort will be required to provide suitable systems for acquisition and quantity guaging of LOX and LH_2 under the zero-g conditions existing during propellant transfer to SOC. Adequate acquisition and guaging techniques are available but some zero-g hardware testing is still required.

4.11 SHUTTLE TRAFFIC MODEL

In addition to the preceding analysis of specific techniques for the basic SOC resupply functions such as logistics module exchange procedures, propellant transfer, etc., the need exists to develop a general feel for the amount of SOC related traffic likely to appear. Traffic levels are needed to assist in system sizing trades and to determine the crossover thresholds for introducing new functions and/or facilities into the overall SOC operating scenario. Traffic levels can also affect shuttle turnaround operations and related ground facilities as well as the need for Shuttle improvements in delivery performance.

Thus, the objective of this task is to develop a preliminary shuttle traffic model using SOC logistics requirements generated by the SOC System Definition Contractor (Boeing Aerospace Company) and to then analyze the main Shuttle interactions for their potential impacts on the Shuttle and related STS facilities. Among the key questions and issues of interest are the following:

(1) Should a dedicated Shuttle be applied to support SOC operations?

(2) What are the effects of Shuttle turnaround time?

(3) What should be a fleet utilization policy or philosophy?

(4) How would the fleet utilization policy be affected by OTV propellant storage on SOC? Also, how would scavenging of fuel from the Shuttle external tank (ET) affect flight packaging, fleet utilization and overall costs and schedules?

(5) How do the SOC and MOTV crew rotation requirements affect fleet utilization policy?

These are the main Shuttle related questions. It is anticipated that Boeing will further analyze the traffic implications for their impacts on the SOC configuration and its growth requirements.

4.13.1 Traffic Model Construction

A preliminary Shuttle traffic model for SOC operations was constructed by evaluating and adjusting raw payload and manifest data provided by Boeing Aerospace Company. Adjustments were made to rule out several payloads which either had physical incompatibilities with the orbiter bay or required high orbit inclinations not compatible with the SOC at 28.5 degrees. Further adjustments were applied to assure bay length allowances reflected the installation of a docking module (7 feet of bay length) on all SOC supply missions and to provide additional Shuttle flights to return OTV stages for ground refurbishment after 8 mission cycles. Raw data features and the specific nature of the adjustments are discussed in the following paragraphs.

Input data showing yearly occurrences for each payload and/or flight category during the years 1985 to 2000 are shown in Table 4.3. The corresponding estimated number of Shuttle flights per year for ground-based and space-based OTV concepts are presented in Figure 4.40.

The corresponding total cargo mass per year delivered to SOC was determined. Figure 4.41 and Table 4.4 present these estimates of total required payloads to SOC for each year, in terms of net useful mass (special support equipment and tank weights not included). These data were derived from the original Boeing-supplied computer printout, in order to expedite the schedule of analysis work. The total fuel mass is essentially equal to that in the revised version, but varies a small amount within given years.

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TABLE 4.3 PAYLOADS SCHEDULES AND DESCRIPTIONS (Page 1 of 3)

NO.	KEY	PAYLOAD DESCRIPTION	MANIFEST RESTRICT	87	88	89	90	91	92	93	94	95	96	97	98	99	00	MASS UP	MASS DOWN	LENGTH	DELTA V UP	DELTA V DOWN	DELTA V A/B	CONSTRUCTION
				TRAFFIC MODEL YEAR																				
1	NONE	SOC CREW ROT. AND RESUPPLY	0	4	4	4	4	4	4	4	4	4	4	4	4	4	4	20.00	15.00	8.00	0	0	0	NO
2	1	GEO PLATFORM DEMONSTRATION	0	0	1	0	0	0	0	0	0	0	0	0	0	0	0	5.67	0.00	7.62	4398	4412	2210	YES
3	2A	GEO PLATFORM COMMERCIAL	0	0	0	0	0	1	1	1	0	1	1	0	0	0	0	6.80	0.00	7.62	4398	4412	2210	YES
4	2A	ADVANCED GEO PLATFORM	0	0	0	0	0	0	0	0	0	2	2	4	2	4	4	14.74	0.00	15.24	4398	4412	2210	YES
5	2B	AUTO. GEO PLAT. SERVICING	0	0	0	0	0	0	0	1	1	1	1	1	1	1	1	3.49	3.49	7.62	4398	4412	2210	NO
6	2B	MANNED GEO PLAT. SERVICING	0	0	0	0	0	0	0	0	0	0	1	1	1	1	1	7.80	7.80	10.67	4398	4412	2210	NO
7	3	PLANETARY MARS AUTO ROVER	0	1	0	0	0	0	0	0	0	0	0	0	0	0	0	5.41	0.00	7.62	3582	3914	652	NO
8	3	PLANETARY SOLAR POLAR	0	0	1	0	0	0	0	0	0	0	0	0	0	0	0	1.79	0.00	7.62	7745	0	0	NO
9	3	PLANETARY SOPP+SEPS	0	0	0	1	0	0	0	0	0	0	0	0	0	0	0	4.70	0.00	7.62	3889	4528	959	NO
10	3	PLANETARY IOO	0	0	0	0	0	1	0	0	0	0	0	0	0	0	0	3.67	[illegible]	7.62	4355	5460	1425	NO
11	3	PLANETARY J/M FLYBY	0	0	0	0	0	0	1	0	0	0	0	0	0	0	0	1.02	0.00	6.00	6911	0	0	NO
12	3	PLANETARY MEO	0	0	0	0	0	0	0	1	0	0	0	0	0	0	0	5.06	0.00	7.62	4314	5378	1384	NO
13	3	PLANETARY URANUS O/P	0	0	0	0	0	0	0	0	1	0	0	0	0	0	0	1.95	0.00	6.00	4845	6440	1916	NO
14	3	PLANETARY MARS SSR	0	0	0	0	0	0	0	0	0	1	0	0	0	0	0	4.37	0.00	7.62	3671	4092	741	NO
15	3	PLANETARY UNSPECIFIED	0	0	0	0	0	0	0	0	0	0	1	1	0	1	0	4.51	0.00	7.62	5544	0	0	NO
16	4	MULTIPLE PAYLOAD DELIVERY	0	2	1	0	2	0	1	0	1	0	1	0	1	0	1	4.76	0.77	7.62	2400	2415	340	NO

TABLE 4.3 PAYLOADS SCHEDULES AND DESCRIPTIONS (Page 2 of 3)

NO.	KEY	PAYLOAD DESCRIPTION	MANIFEST RESTRICT	TRAFFIC MODEL YEAR 87	88	89	90	91	92	93	94	95	96	97	98	99	00	MASS UP	MASS DOWN	LENGTH	DELTA V UP	DELTA V DOWN	DELTA V A/B	CONSTR-UCTION
17	5	DOD CLASS 1A GENERIC	1	1	2	2	1	2	2	2	2	2	2	2	2	2	2	3.18	0.00	6.10	4398	4412	2210	NO
18	6	DOD CLASS 1B	1	1	1	1	1	1	1	1	1	0	0	1	1	1	1	2.27	0.00	12.80	4398	4412	2210	NO
19	7	DOD CLASS 2 GENERIC	1	0	0	1	2	2	2	1	0	0	0	0	0	0	0	5.44	0.00	6.10	4398	4412	2210	NO
20	8	DOD CLASS 3 GENERIC	1	0	0	0	0	1	0	1	1	2	1	2	1	2	1	11.34	0.00	7.62	4398	4412	2210	NO
21	9A	SPACE RADAR POLAR	2	0	1	1	2	2	1	1	0	0	0	0	1	1	2	4.54	0.00	10.00	0	0	0	NO
22	9B	SPACE RADAR GEO	0	0	0	0	0	0	1	1	0	0	0	0	0	0	0	9.07	0.00	18.20	4398	4412	2210	YES
23	10	PERSONAL COMMUNICATION	0	0	0	0	0	0	0	1	0	1	0	1	0	1	0	8.16	0.00	18.20	4398	4412	2210	YES
24	11	AUTOMATED SERVICE AT GEO	0	0	0	0	1	0	1	2	2	3	3	4	3	0	0	2.72	0.91	6.10	4538	4412	2210	NO
26	12B	MANNED SORTIE TO GEO (HI MOD)	0	0	0	0	0	0	1	1	1	2	2	2	0	0	0	6.33	6.33	4.57	4900	4412	2210	NO
27	13	DEBRIS REMOVAL AT GEO (MANNED)	0	0	0	0	0	0	1	2	2	2	2	1	0	0	0	2.87	2.87	3.65	4795	4412	2210	NO
28	14-1	MANNED GEO STATION DEL#1	0	0	0	0	0	0	0	0	0	0	1	0	0	0	0	9.07	0.00	12.20	4398	4412	2210	NO
29	14-2	MANNED GEO STATION DEL#2	0	0	0	0	0	0	0	0	0	0	1	0	0	0	0	9.07	0.00	12.20	4398	4412	2210	NO
30	14-3	MANNED GEO STATION DEL#3	0	0	0	0	0	0	0	0	0	0	1	0	0	0	1	4.54	0.00	12.20	4398	4412	2210	NO
31	15	MANNED GEO STATION SUPPORT	0	0	0	0	0	0	0	0	0	0	2	4	4	4	4	7.48	4.08	7.62	4398	4412	2210	NO
32	16	UNMANNED LUNAR ORBIT SORTIE	0	0	0	0	0	0	1	0	0	0	0	0	0	0	0	1.14	0.00	4.57	4000	4000	1220	NO
33	17	MANNED LUNAR ORBIT SORTIE	0		0	0	0	0	0	0	0	1	0	0	1	0	0	6.07	5.96	12.80	4000	4000	1220	NO

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TABLE 4.3 PAYLOADS SCHEDULES AND DESCRIPTIONS (Page 3 of 3)

NO.	KEY	PAYLOAD DESCRIPTION	MANIFEST RESTRICT	87	88	89	90	91	92	93	94	95	96	97	98	99	00	MASS UP	MASS DOWN	LENGTH	DELTA V UP	DELTA V DOWN	DELTA V A/B	CONSTRUCTION
34	18	DOD CLASS 4	1	0	0	0	0	0	0	0	0	0	2	2	2	2	2	13.60	0.00	18.00	4398	4412	2210	YES
35	19	PINHOLE X-RAY TELESCOPE	0	0	0	0	0	0	0	0	0	0	0	1	0	0	0	16,50	0.00	10.00	3800	3800	1000	YES
36	20	SPS DEMO ARTICLE #1	0	0	0	0	0	0	0	0	0	1	0	0	0	0	0	3.40	0.00	12.00	4398	4412	2210	YES
37	20	SPS DEMO ARTICLE #2	0	0	0	0	0	0	0	0	0	2	0	0	0	0	0	15.50	0.00	18.00	4398	4412	2210	YES
38	NONE	SEGMENTED-MIRROR IR TELESCOPE	0	0	1	0	0	0	0	1	0	0	0	0	0	0	0	25.00	0.00	16.10	0	0	0	YES
39	NONE	LARGE-APERTURE INSTRUMENT	0	0	0	0	0	0	0	0	0	0	4	0	0	0	0	25.00	0.00	16.10	0	0	0	YES

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4.4 MASS SUMMARY - PAYLOADS TO SOC ORBIT (SPACE-BASED OTV)

Year of Flight	SOC Payload (tonnes)	Fuel (tonnes)*			Other (tonnes)	Adjustment (tonnes)	Total (tonnes)
		Scavenger	Tanker	Total			
1987	100.38	45.6	82	127.6			227.98
1988	130.39	81.8	82	163.8	4.17[a]	-4.54[c]	293.82
1989	103.31	78.2	41	119.2	4.17[b]	-4.54[c]	222.14
1990	117.65	82.9	123	205.9	4.17[b]	-9.08[c]	318.64
1991	130.40	93.4	164	257.4	4.17[b]	-9.08[c]	382.89
1992	138.76	40.9	410	450.9	8.34[b]	-10.87[c,d]	587.13
1993	185.04	81.8	492	573.8	8.34[b]	-10.87[c,d]	756.31
1994	127.68	61.9	451	512.9	8.34[b]	-6.33[d]	642.59
1995	228.37	55.	861	916.	12.51[b]	-12.66[d]	1,144.22
1996	345.97	68.	1025	1093.	12.51[b]	-20.46[d,e]	1,431.02
1997	294.74	43.4	1230	1273.4	16.68[b]	-20.46[d,e]	1,564.36
1998	221.39	80.7	779	859.7	16.68[b]	-12.34[c,e]	1,084.43
1999	255.92	100.6	779	879.6	16.68[b]	-12.34[c,e]	1,139.86
2000	245.72	132.1	697	829.1	12.51[b]	-16.88[c,e]	1,070.45
Total				8261.3			10,865.84

NOTES

a. OTV replacement for expended OTV

b. Added flights to Boeing data for refurbishment of OTV; 4.17 = 1 flight in this summary

c. Space radar polar does not go to SOC (deleted from payloads to SOC)

d. Manned sortie to GEO (HI-MOD) not feasible with selected OTV model (deleted from payloads to SOC)

e. Manned GEO platform servicing not feasible with selected OTV model (deleted from payloads to SOC)

* Fuel scavenging and tanker proportions in given year are affected by actual number of SOC payload flights and permissible size of payload. Total is not affected.

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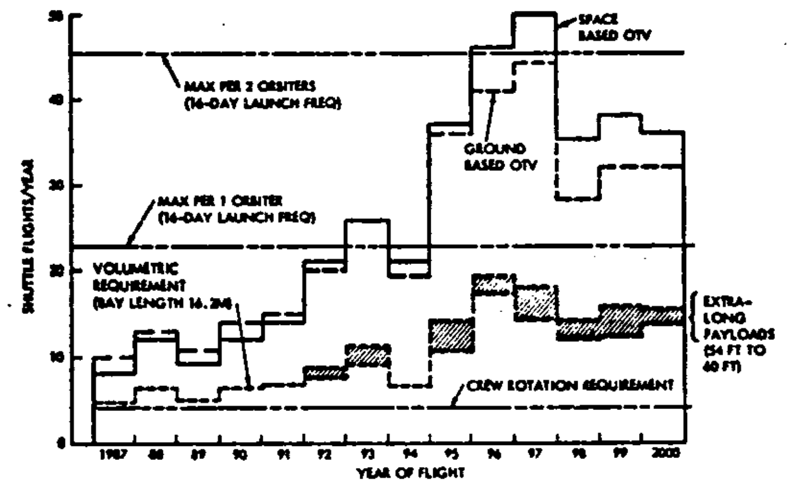

FIGURE 4.40 PRELIMINARY BOEING SUMMARY SOC TRAFFIC MODEL - FLIGHTS PER YEAR

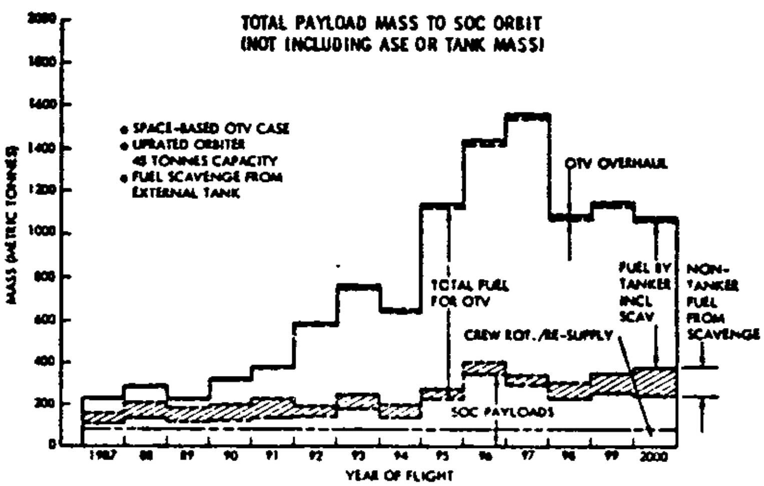

FIGURE 4.41 MASS SUMMARY

Figure 4.41 includes an approximate breakdown of mass to SOC orbit contributed by (1) the actual payloads to SOC (including the SOC crew rotation/resupply), (2) the fuel required to supply OTV flights and (3) the mass transport required for periodic OTV overhaul flights (every eight flights of OTV reuse). One flight (up and down) was allocated for each OTV overhaul, on the assumption that an extra OTV vehicle is available to replace the vehicle being overhauled. In counting such flights, it was assumed that one of the OTVs at the SOC was used for an expendable OTV prior to accumulating eight flights. Then, this OTV mission would be followed by a shuttle flight bringing up a replacement OTV which could accomplish eight flights prior to overhaul. Note that the fuel mass was estimated by summing the totals provided by tankers and scavenging operations for each year. There is some carryover from year to year. To help understand the basic makeup of these flights, Figure 4.40 also shows the estimated total volume of all delivered payloads in terms of (1) crew rotation flights and (2) equivalent orbiter flights required merely to meet the total cumulative length requirements of all payloads delivered each year. The latter was calculated assuming an available bay length of 16.2 meters (53 feet), so as to include a docking module on all flights. The remaining flights would be required to account for OTV propellants, OTV replacement and unusable empty volume on flights where insufficient volume was available to accommodate the next payload on the manifest list.

In performing this analysis it was noted that certain flights required bay lengths of up to 18.2 meters (60 feet); that is, no docking module could be accommodated. The minimum number of flights potentially affected is indicated by shaded areas. Combinations of two or more payloads with lengths greater than 16.2 meters are not separately broken out. A revised computer analysis of shuttle manifests was obtained from Boeing which incorporated the volume and mass restrictions of the docking module. These manifests also account for data not included in Table 4.3, such as OTV vehicles and tanker payloads destined to the SOC which are required to transport payloads to GEO or other orbits beyond the SOC.

Included in Fig. 4.41 is a shaded area indicating one example of how much of the OTV fuel could be garnered by scavenging from the External Tank during ascent on non-tanker flights. The actual amount of such scavenging is a function of the specific payloads characteristics and orbiter lifting capability. In turn, these factors are related by possible matching of payloads and scavenging tanks according to length constraints, mass and other compatibility constraints, to yield a total number of orbiter flights. The specific example shaded area shown was derived from the preliminary Boeing traffic model data. Although not included in the shaded zone in the figure, ET propellant scavenging was applied to the tanker. Two sizes of scavenge tanks were assumed, one occupying 1.5 meters of cargo bay length and having 4.7 tonnes mass capacity, and the other having length of 3 meters and 10.5 tonnes capacity.

The main purpose of Figure 4.41 is to characterize the magnitude of the total mass transport problem, not to examine in detail all of the possible implications of scavenging systems and payload matching interactions. Such detailed interactions are deferred to future SOC studies. In addition to the mass of SOC-related payloads listed above, each orbiter flight will carry support equipment which is payload-chargeable to the orbiter flights, such as docking modules, scavenging tanks, PIDAs or HPAs, lights, etc. with penalties as listed in Table 4.5. Such items may reduce the usable effective cargo bay volume and launch mass carrying capability of the orbiter for SOC payloads. However, the scavenge tank set has the effect of increasing the effective launch load carrying capability while reducing the available cargo bay length for launch. If these support equipment items are not required on some flights because of the nature of the payload or offloading operations, increased volume and mass could be transported.

Applying the foregoing considerations to the yearly mass delivery requirements, the preliminary flight rate data of Figure 4.40 were adjusted to the results presented in Figures 4.42 and 4.43 . Figure 4.42 shows the estimated minimum number of Shuttle flights per year based on the standard orbiter which has 29.5 tonnes lift capacity and 16.2 meters of cargo bay length available (behind the docking module).

Figure 4.43 is based upon the use of an uprated orbiter of 45 tonnes lifting capacity, also with 16.2 meters of cargo bay length available. Both Figures 4.42 and 4.43 assume a space-based OTV. Both figures show the breakdown of the number of flights allocated to SOC crew rotation and re-supply, SOC payloads, fuel tanker flights for OTV support, and an estimate of flights required to refurbish and overhaul OTVs. The SOC payloads include replacement OTVs, (never more than one in a given year).

To aid in understanding the level of space activity supported by these flight rates Table 4.6 was prepared. This table summarizes the number of payloads per year for each of 7 basic mission categories. However, these preliminary mission data and traffic characteristics represent just the beginning. Much mission definition effort remains to be done in the future. Contacts must be made with the user community to confirm, update or establish the mission needs and to develop meaningful forecasts of market growth for the services and products to be produced. These in turn must be synthesized into suitable mission and project concepts. Although much remains to be done, these preliminary data give some correlation between projected levels of space activity and the Shuttle traffic required to support it.

TABLE 4.5 POTENTIAL SUPPORT EQUIPMENT PENALTIES

ITEM	‡ USE P	T	PENALTY MASS tonnes	lb	LENGTH meters	ft	SAVINGS MASS* tonnes	lb
• DOCKING MODULE	X	X	1.769	3900	2.06	6.75		
• SCAVENGE TANKS (BOEING)	X X		(NOT SPECIFIED)		1.5 3.0	4.92 9.84	4.7 10.5	10,363 23,152
• SCAVENGE TANKS (ROCKWELL)	X		0.363	800	2.87	9.42	10.7	23,600
• SCAVENGE PLUMBING (ROCKWELL)	X	X	0.091	200	N/A	N/A		
• PIDA (2)	X		0.181	400	N/A	N/A		
• HPA (1)	X		0.158	350	N/A	N/A		
• TANK FOR TANKER FLTS		X	8 TO 9	18 TO 20K	(NOT DRIVER)		(SIZED FOR SCAVENGE)	
• OMS KIT (EMPTY)	X	X	1.35 TO 2.39	2978 TO 5275	2.87	9.42		
(FULL)	X	X	2.39 TO 19.23	15,380 TO 42,408	2.87	9.42		
• LIGHTS	X	X	(TBD)		(TBD)			
• TV CAMERAS	X	X	(TBD)		NEGLIGIBLE			

*INCREASES ON-ORBIT DELIVERY MASS (NOT PENALTY)

‡P = PAYLOAD FLIGHT
T = TANKER FLIGHT

TABLE 4.6 MISSION CHARACTERISTICS, SOC-DESTINED SHUTTLE FLIGHTS

	YEAR OF FLIGHT / NUMBER OF FLIGHTS 1987	1988	1989	1990	1991	1992	1993	1994	1995	1996	1997	1998	1999	2000
• PLANETARY & LUNAR MISSIONS		1	1		1	1	1	1	2	1	1		1	
• GEOSTATIONARY PLATFORM DEL. & SERVICE INCL. GEO RADAR	1	1		1	1	3	6	5	8	8	11	7	4	5
• GEOSTATIONARY SPACE STATION (MANNED) DELIVERY & SERVICE MISSIONS										5	4	4	4	5
• DEPT. OF DEFENSE MISSIONS	2	3	4	4	6	5	5	5	2	5	5	4	7	6
• MISC. MULTIPLE PAYLOADS	2	1		2		1		1		1	1	1		1
• DEBRIS REMOVAL (GEO & OTHER ORBITS)						1	2	2	5	2	1			
• LARGE, SPECIAL PAYLOADS DELIVERY & DEPLOYMENT						1	2	2	5	2	1			

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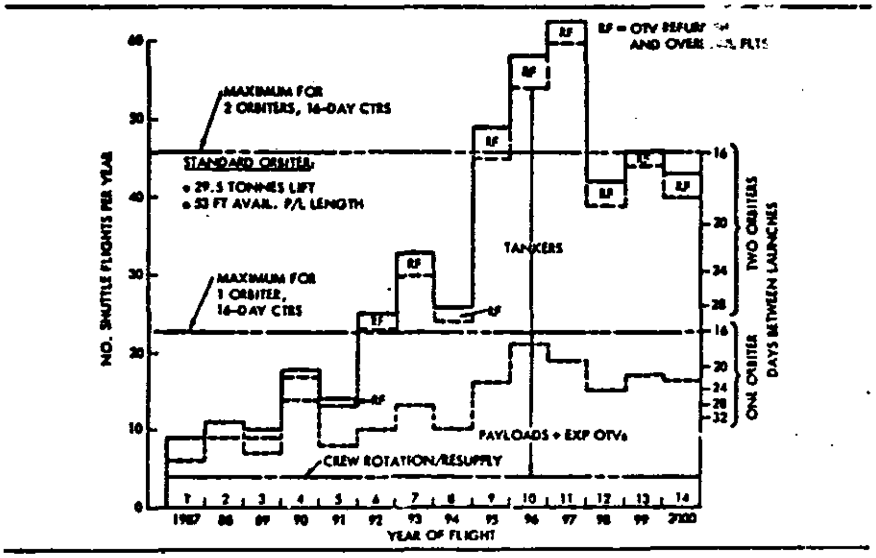

FIGURE 4.42 TRAFFIC MODEL, SOC-[illegible] ORBITERS
STANDARD SHUTTLE

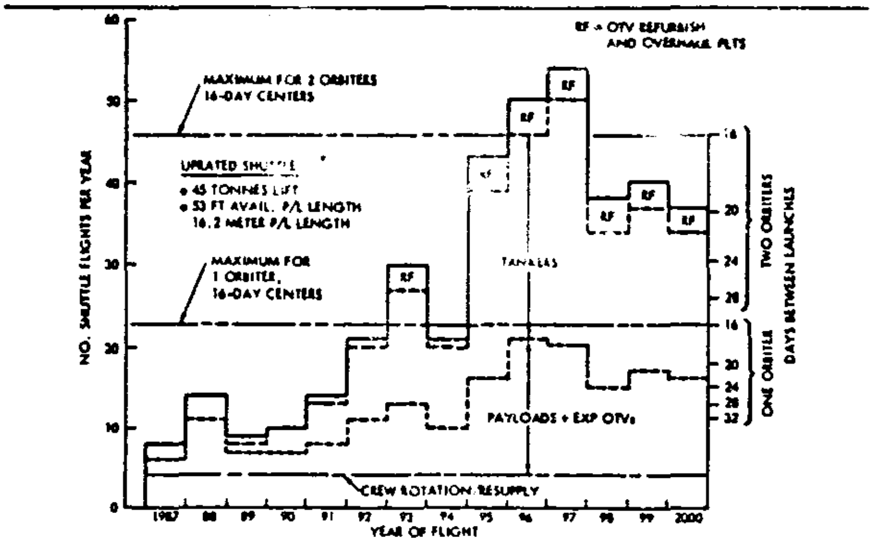

FIGURE 4.43 TRAFFIC MODEL, SOC-DESTINED ORBITERS
UPRATED SHUTTLE

4.11.2 Analysis of Traffic Impacts

Dedicated Orbiter

In light of the high traffic levels predicted, one of the key questions which arises is the possibility of dedicating one or more orbiters to SOC flights. The greatest value of such an arrangement is that ground turnaround time (and thus costs) could be reduced. In addition, there is a potential gain in reliability, since there are fewer requirements for making and breaking electrical connections and fluid connections, less potential for ground handling damage of parts and fewer logistics and stocking problems. These gains are based on the presumption that many of the same equipment items will be carried on all (or nearly all) SOC flights. Such potential equipment items are listed in Table 4.7. In actual operations, some items may be removed for savings of weight or volume, but the majority would still be aboard all SOC-destined flights.

In addition, most SOC-destined flights could omit standard sleep stations, as well as some food and storage supplies.

Another option in this regard is the airlock. With a dedicated orbiter having committed the volume for a docking module on all flights, it could well be advantageous to make the docking module also serve as an airlock. The current multi-purpose docking module concept established during the SOC study would probably require a separate kit of ECLSS equipment located in the crew cabin for pressure suit donning and doffing. This equipment would be mounted in part of the volume left by removal of the standard airlock. The crew preparing to go out for EVA would don suits in the cabin, then egress through the airlock. As noted in Table 4.7, the weight savings could be approximately 905 lbs.

The dedicated orbiter concept probably would not be cost effective early in the SOC program, since few flights per year would be required. The optimum point of introduction would be near the time when total traffic per year is sufficient to require all the flights which can be generated by one orbiter with the expected turnaround time. The longer the required turnaround time, the sooner the orbiter should be dedicated to SOC operations. Eventually, traffic could build up to the point where two (or more) dedicated orbiters could be effectively used. If flight traffic to the SOC exhibits large up and down variations in succeeding years, it may be desirable to return a dedicated orbiter to general service operations for a time, until the expected traffic to the SOC rises again.

If it becomes desirable to dedicate two orbiters to the SOC traffic, one may be set up exclusively for tanker flights (plus some crew rotation). This option is attractive because the number of tanker flights in later years tends to equal the number for other purposes. Certainly, the installation and removal of a fuel tank involves a significant time constraint in turnaround time, because of the mass, the bulk and the plumbing involved. The tankers could omit PIDAs and probably HPAs to enhance load carrying capability. However, installation of tanks without removing such equipment might be considered as a time-saving feature for turnaround operations, especially in earlier years when both tanker and cargo flights could be supported with one orbiter.

4.7 EQUIPMENT CONSIDERATIONS FOR A SOC-DEDICATED ORBITER

EQUIPMENT ITEMS	ADD/ OMIT/ MOD	GROUND TURNAROUND SAVINGS (OPERATIONS)	COST IMPACTS	WEIGHT ESTIMATE tonnes	WEIGHT ESTIMATE lb
• DOCKING MODULE (DM)	ADD	OMIT INSTALLATION, C/O & REMOVAL TIME	INCREASE	+1.8	+[illegible]
• STANDARD AIRLOCK	OMIT	POSSIBLE REMOVAL OF STANDARD AIRLOCK & PROVIDING CAPABILITY IN DOCKING MODULE —NO TIME SAVINGS; NEED KIT FOR SUIT DON/CHECKOUT/DOFF EQUIP. IN CABIN	ONE CHANGEOUT PER DEDICATED ORBITER; SLIGHT INCREASE	-0.41 (AIRLOCK SHELL WT)	-905
• ECLSS/SUIT CHECKOUT D&C (KIT)	ADD			(NO CHANGE)	
• PIDA (2) (EXCEPT ON TANKER FLIGHTS?)	ADD	OMIT INSTALLATION, CHECKOUT, AND REMOVAL TIME	INCREASE	+0.18	+400
• HPA (EXCEPT ON TANKER FLIGHTS?)	ADD		INCREASE	+0.16	+350
• ΔLIGHTS IN CARGO BAY, DM	ADD		INCREASE	(MINOR CHANGE)	
• ΔCAMERAS IN CARGO BAY, DM	ADD		INCREASE	(MINOR CHANGE)	
• AFD CONSOLE	MOD	MINIMIZE INST., C/O & REMOVAL TIME	MINIMAL	(MINOR CHANGE)	
• SEATS & RELATED EQUIP. —MID-DECK	ADD (TBD)	MINIMIZE LOGISTICS CONTROL OPERATIONS. POSSIBLE SCAR WT PANELS IN FLOOR STRUCTURE FOR 8-MAN SOC RESCUE	(INCREASE TBD)	(MINOR CHANGE)	
• SEATS—CREW STA. AFD	N/C	MINIMIZE LOGISTICS CONTROL OPERATIONS	(NO CHANGE)	(NO CHANGE)	
• GALLEY	MOD	MINIMIZE LOGISTICS CONTROL, REFURB.	MINIMAL	(MINOR CHANGE)	
• SLEEPING STATIONS VS. SLEEPING BAGS	OMIT	POSSIBLY ELIMINATE INSTALLATION TIME OF SLEEP STA. ADD SLEEPING BAG STOWAGE, ATTACH PROVISIONS	DECREASE	-0.06	-128
• DEDICATED STOWAGE BOXES	MOD	MINIMIZE LOGISTICS CONTROL, REFURB.	DECREASE	(MINOR CHANGE)	
• RMS	N/C OR OMIT	USE ON ALL MISSIONS IS LIKELY; NO IMPACT ON INSTAL./REMOVAL. (POTENTIAL 1-TIME REMOVAL/REPLACEMENT FOR DEDICATED TANKER)	NO CHANGE (TBD)	(NO CHANGE) (-0.5)	(-1000)
• SCAVENGE/TRANSFER SYSTEM FOR ET FUEL	ADD	DESIGN NOT DETERMINED; POSSIBLY SOME SCAR, SOME REMOVABLE ITEMS WHICH WOULD NOT BE HANDLED	INCREASE	+0.09	+200
• ELEC. HARNESS IN CARGO BAY (SPECIAL ITEM FOR SOC FLIGHTS)	MOD OR ADD	OMIT CHANGES. TBD DIFFERENCE IN WT., ACCESSIBILITY VS. SMCH	MOD. INCREASE	SMALL DECREASE (TBD)	
• RENDEZVOUS EQUIPMENT	N/C	(TBD) MAY BE STANDARD FOR MANY OTHER TYPES OF FLIGHTS	NO CHANGE	NO CHANGE	
• THIRD CRYO TANK SET FOR ELEC. POWER	OMIT	ONE REMOVAL & REPLACEMENT PER DEDICATED ORBITER	(TBD)	-0.68	-1500

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The potential need and timing for introducing dedicated orbiters was explored for the traffic projections in Section 4.11.1. Previous Figures 4.42 and 4.43 provide indications of how many orbiters could be required per year to support the traffic to the SOC, based on differing estimates of average total time for each flight and its associated ground turnaround time.

The alternating long-and-short dashed lines extending horizontally across the graphs show the maximum number of flights for one or two orbiters per year based upon a 16-day interval between launches (14-day turnaround and 2-day mission). Where such lines cut across the vertical bars, the traffic is presumed adequate to support and justify an orbiter dedicated exclusively to the SOC-destined traffic. Along the right side of each graph is a scale indicating the effects of increased turnaround time on the number of yearly flights achievable with a single orbiter. Utilizing these scales the points of intersection with vertical bars were determined, and the resulting trends are presented in Fig. 4.44 for the standard orbiter. Thus, Figure 4.44 presents estimates of the time when one might introduce dedicated orbiters for SOC flights. The graph shows the number of years from the beginning of SOC full-up operations (after 1986 in this model) until various additional dedicated orbiters could be utilized for a period of at least one full year. Also indicated are shaded areas representing conditions in which one might wait until later, when two or more full years of operation with a dedicated orbiter could be expected. For example, if the average time between launches amounts to 24 days, as many as four dedicated orbiters could be utilized in peak traffic years. Figure 4.44 does not attempt to trace each case of possible return to general service operations when peak traffic drops off. These data are based on the use of a standard orbiter. A similar graph for the uprated orbiter (not shown) would, in general, show longer times for introduction of dedicated orbiters, with little need for a fourth dedicated orbiter.

Turnaround Time

Figures 4.42, 4.43 and 4.44 showed potential influences of turnaround time on introduction of the SOC-dedicated orbiter and number of shuttle flights per year which could be accomplished by a given orbiter. These considerations interact with SOC dedication concepts and total SOC/orbiter fleet utilization. Potentially, when a SOC-dedicated orbiter goes into service there is a temporary improvement in fleet usage capability for the remaining orbiters, since the SOC-dedicated orbiter should be capable of accomplishing more flights per year due to lower ground turnaround operations.

On the other hand, if orbiter turnaround time cannot be held to a minimum, more orbiters would be needed to accomplish the missions listed in the time scheduled, and these orbiters would be required earlier as the turnaround time is lengthened. In actuality, the turnaround time should tend to decrease as experience is gained. Thus, a line showing average capability for a given orbiter over the years would have a slope upward to the right, more nearly paralleling the increase in traffic. Such a trend would more strongly favor early dedication of an orbiter to SOC-destined traffic.

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Fleet Utilization Policy

The foregoing discussion provides several insights related to orbiter fleet utilization policy. For example, it appears that there will be an optimum time to begin utilizing one or more orbiters exclusively for SOC operations after the program gets well underway. The timing is strongly dependent on the ground turnaround time and somewhat related to SOC stay time. Other considerations involving equipment modifications for scavenging external tank fuel could also drive toward early dedication of at least one orbiter so as to minimize the number of tanker flights.

OTV Propellant Storage on SOC

Among the concerns listed early in this section were two questions relating fleet utilization to storage of propellant on the SOC and scavenging of fuel from the ET. The preliminary Boeing data provided a comparison of number of shuttle orbiter flights required to support space-based OTVs and ground-based OTVs. As shown in Figure 4.40, the Boeing data suggests that more orbiter flights would be required for ground-based OTVs early in the SOC program (first five years), but fewer would be required later. Presumably, this effect is largely due to the reduced fuel requirements of OTVs returned by aerobraking techniques. The uprated shuttle requires fewer flights than the standard shuttle to deliver the required OTV propellant to SOC. However, in both cases a significant number of additional flights are required if the OTVs are returned to earth for refurbish/overhaul after every eight flights. Boeing analysis has indicated the period between overhauls could potentially be extended to about twenty flights, which would significantly reduce the refurbish/overhaul flights.

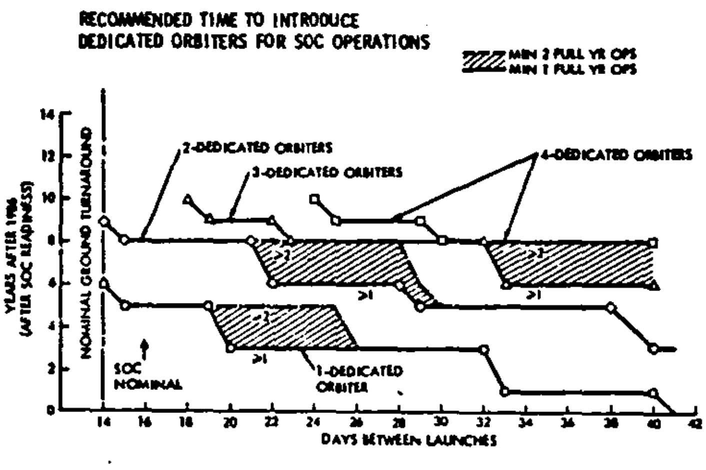

FIGURE 4.44 DEDICATED ORBITER INTRODUCTION STANDARD ORBITER

Although it is beyond the scope of this study, there are a number of OTV sizes and concepts which must be analyzed to determine the most effective operational approach and design for use with the SOC. Significant differences in OTV propellant requirements are possible for the wide range of OTV concepts which can be considered. However, propellant will still be the largest single category of SOC cargo. Hence, the ability to store propellants on SOC is very important to the overall logistics efficiency. It allows propellant scavenging from the ET which reduces the number of tanker flights. In conjunction with space basing it eliminates OTV size constraints over ground based OTV designs which must be delivered to orbit while full of propellant and thus would be size limited to the 65K lb shuttle performance limit. Propellant storage also uncouples propellant delivery schedules from specific mission schedules thereby easing fleet management problems. Propellant can be delivered on a routine basis rather than in clusters of launches to support a given mission.

The specific effect on orbiter flights of scavenging ET fuel includes some use of cargo bay volume for the fuel transferred from the external tank. Boeing analyses assumed two possible tank sizes - 1.5 meters long and 3.0 meters long. Most of the combined payloads and single payloads could accommodate one or the other of these tanks. As a result, sufficient fuel was scavenged to save 2 to 3 tanker flights per year with a standard shuttle and one to two tanker flights per year for the uprated shuttle (45 tonnes capacity). In addition, tanker flights themselves could also deliver more fuel to orbit than that carried in the cargo bay at launch. The Boeing computer printout did not separately identify the savings associated with scavenging on these flights, although it was included in the total propellant delivery calculations. With the number of tanker flights roughly equal to the number of payload delivery flights in the latter years, it can be estimated that equivalent savings could be expected due to scavenging, that is, 4 to 6 flights for standard shuttles and 2 to 4 flights for uprated shuttles.

4.11.3.5 Crew Rotation Effects

The available data for this traffic model analysis did not have sufficient granularity to make significant analyses of crew rotation effects on fleet utilization policy. However, it can be inferred that the projected large number of flights, after approximately the first 5 years, would allow a considerable flexibility of options in crew rotation scheduling. At flight rates of 40 to 50 per year, the OTV crew would seldom have more than one week to wait before they could be brought up to the SOC and/or returned to earth. The conclusion from this brief consideration is that SOC and MOTV crew rotation requirements would have little or no effect on fleet utilization policy.

Traffic Model Data Sources

1. Boeing

   Unpublished copy of computer printouts - Payload Descriptions and Shuttle Traffic Model, Ground-Based and Space-Based OTV (Preliminary). Received March 3, 1981.

2. Boeing

   Unpublished computer printout - Shuttle Traffic Model for Space-Based OTV Standard and Uprated Orbiter Received March 17, 1981.

3. MSFC

   Internal Memorandum - Preliminary OTV Mission Model - Revision 2, Feb. 27, 1980.

References

1. Orbital Propellant Handling and Storage Systems Definition Study Final Report, Vol. II Technical; General Dynamics, Convair Division, Report No. GDC-ASP-79-002, dated 15 August 1979; Contract NAS9-15640

2. In-Space Propellant Logistics and Safety, Vol. II Technical: North American Rockwell, Space Division; Report No. SD72-SA-0053-1, dated 23 June 1972; Contract NAS8-27692, NASA/MSFC

3. A Study of Hydrogen Slush and/or Hydrogen Gel Utilization, Final Report; Lockheed Missiles and Space Company, Report No. K-11-67-1; dated 11 March 1967; Contract NAS8-20342, NASA/MSFC

4. NHB 1700.7 "Safety Policy and Requirements for Payloads Using the Space Transportation System (STS)", NASA/JSC May 1979

5. MMS In-Orbit Refueling Study Analysis and Design; Rockwell International, Space Operations and Satellite Systems Division; Report No. SSD80-0175-1, dated 4 December 1980; Contract NAS5-26152, NASA/GSFC

6. Space Construction Systems Analysis Study, NAS9-15718 April 1980, Document SSD 80-0038, Rockwell International

5.0 FLIGHT SUPPORT FACILITY

Three principal objectives are identified for this task; (1) determine the implications to the SOC to perform the activities associated with spacecraft assembly, launch, recovery and servicing, (2) determine the implications to the Shuttle to perform these same activities, and (3) to determine the unique requirements imposed on the space vehicle to permit these space based activities.

This section is composed of the subjects of the servicing philosophy associated with space based servicing activities, the SOC arrangement concept providing these services; the Shuttle implications when performing similar functions before SOC is available and the unique requirements imposed on the spacecraft to permit space based servicing.

5.1 SUMMARY

Servicing Philosophy

A servicing philosophy is developed that addresses servicing of an MOTV/OTV. In general, the philosophy is modeled after airline type practice where maintenance (LRU replacement) is performed only as a result of indicated deteriorating performance, failure of redundancy, wear, or when the time/cycle of critical equipment is scheduled for replacement. This philosophy avoids the replacement of a good operating unit when its performance history does not warrant it. The incorporation of warehousing facilities on the SOC to store spare parts principally for planned replacement, but also for unplanned critical parts was identified. The concept of utilizing replaceable system packages, similar to the multi-mission system packages, mounted to the external surface of the space vehicle is recommended. This concept permits removal/replacement by remote or EVA methods.

The philosophy also identified the desirability of providing dedicated positions for the maintenance activities, particularly for the unscheduled activities. The unscheduled positions can also be utilized for the storage of large spacecraft elements. Both the scheduled and unscheduled maintenance positions have the flexibility to be utilized for either function as the flight and maintenance rates dictate. The implementation of this philosophy is represented in the flight support facility configuration concept shown in Drawing 42690-019, Appendix A.

Flexible Multiple Servicing Functions Capability

The preferred servicing facility concept has a hexagonal cross section fixture configuration. This arrangement provides six positions that can be utilized for assembly, staging and servicing operations. Scheduled and unscheduled maintenance stations can be accommodated and storage of large units can also be accommodated with this concept. Multiple mobile servicing manipulators provide access to all six positions.

Growth Capability

A servicing facility growth concept was developed. This concept considers that the early SOC operations will be concerned primarily with the assembly and preparation for launch of single stage OTV's with GEO payloads. The OTV's are designed for space based operations and would return for servicing and preparation for other similiar missions. As these activities increase and include MOTV missions the facility can grow to accept the increased activity. The growth is accomplished principally by adding another section of the facility fixture. Additional growth facilities such as propellant storage can also be added when appropriate. This propellant storage facility is independent of the servicing fixture growth and thus can be added at any phase of the SOC servicing facility growth.

OTV Operations From the Shuttle

Prior to SOC operations, the launch, recovery and servicing of an OTV from the Shuttle may be desireable. The analysis of this type of operation for an OTV designed to operate from the SOC indicated that the characteristics required of the OTV to be serviced from the SOC will permit similar servicing operations from the Shuttle. The Shuttle docking module and RMS provide the holding and maneuvering activities. An adapter to the docking module to hold the OTV for the installation of the payload is identified. The adapter provides the capability to orient the OTV stage to assist in mating the payload. The OTV stage utilizes a standard berthing port at the forward end for mating of the payload. The PIDA attachments on the OTV provide for the controlled deployment from the payload bay and will also provide a mating point to the docking module adapter.

5.2 FLIGHT SUPPORT FACILITY CONSIDERATIONS

Three principal areas of concern that need to be addressed in order to develop the flight support facility and its operations are presented here. The three areas are (1) spacecraft servicing (2) EVA and/or remote manipulator operations and (3) facility growth.

5.2.1 Spacecraft Servicing

The objective of this task is to describe a servicing philosophy from which fundamental requirements may be generated that can be utilized to develop a SOC flight support facility. The servicing philosophy is developed for servicing an MOTV/OTV as a model. Servicing of an MOTV/OTV and other spacecraft are sufficiently similar so that this philosophy should apply to most servicing activities.

The issues considered in developing the servicing philosophy are listed in Table 5.1. An MOTV model derived jointly by Rockwell International, Boeing Aerospace and NASA/JSC also includes items that bear on maintenance/servicing and these items are also considered (Table 5.2).

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TABLE 5.1 SERVICING PHILOSOPHY ISSUES

ISSUES	CONSIDERATIONS
• MAINTENANCE/SERVICING	• SCHEDULED MAINTENANCE • UNSCHEDULED MAINTENANCE
• LEVELS OF REPLACEABLE ITEMS	• LRU COMPONENTS • SUBSYSTEM ASSEMBLIES
• WAREHOUSING	• SPARE PARTS, TOOLS, ETC. • SPECIAL EQUIPMENT
• STORAGE	• CREW MODULES • OTV ELEMENTS • PROPELLANT
• OPERATIONS	• EVA • MANIPULATOR

TABLE 5.2 MOTV MODEL

- UNMANNED OTV MISSIONS WILL OUTNUMBER MANNED MISSIONS BY 3 TO 1
- SOC STOWAGE PROVISIONS FOR MOTV CREW MODULE
- TWO-PERSON MOTV CREW COMPLEMENT
- NO AIRLOCK IN MOTV CREW MODULE
- MOTV TURNAROUND OPERATIONS WILL INCLUDE AN EVA "WALK-AROUND" INSPECTION
- MOTV/OTV RETURN TO EARTH AFTER 8 MISSIONS FOR MAJOR GROUND OVERHAUL
- MOTV/OTV WILL HAVE SELF-DIAGNOSTIC CAPABILITY WITH BUILT-IN COMPUTER SWITCHING TO REDUNDANT UNITS
- CONSIDER TANDEM AND PARALLEL TANKING STAGING ARRANGEMENTS FOR SERVICING AT THE SOC

Maintenance/Servicing

The SOC maintenance/servicing activities approach is modeled along the lines of airline operations. This approach prescribes scheduled maintenance to be performed periodically and unscheduled maintenance to be performed as a result of failure, damage or pending hazardous conditions. A space-based operations flow is depicted in Figure 5.1. The activities that are performed in the scheduled maintenance mode and those performed in the unscheduled mode are indicated in Figures 5.2 and 5.3. Both MOTV and OTV activities are indicated in these figures.

The scheduled flow will include, at the outset, the inspection of the vehicle physically and the subsystem and performance data reviewed. Replacement of time critical and performance degraded LRU's will be performed followed by mechanical and leak checks and subsystem functional checkouts, followed by a complete system checkout. All checkouts will be primarily automatic, controlled by the vehicle onboard computers. The flight support facility will supplement the onboard computer by providing data processing and analysis support. All software analysis support will also be provided in the flight support facility to minimize the software burden on the MOTV/OTV flight computers. The scheduled maintenance operations in the course of normal processing of an MOTV and OTV would replace those LRU's planned for replacement and those necessary to bring the subsystem within operating specifications.

The unscheduled maintenance/servicing activities primarily perform the more difficult and involved repairs. Some examples of the type of servicing and repairs are indicated in Figure 5.3. The unscheduled repair activities are considered more time consuming than the scheduled servicing because of the difficulty or complexity of the operations, and because the required parts may not be readily available to perform the operation. These repairs would only be performed when they were determined to be cost-effective within the mission risk considerations. In order to maintain a given flight rate in orbit it may be desireable to have MOTV/OTV units in storage aboard the SOC. With this capability a spare unit could be processed as a replacement MOTV/OTV. These spare vehicles would most probably be stored as modular units since a total vehicle may not need replacement, but only a modular element such as a propulsion unit or a crew module unit.

Levels of Replaceable Units

Two generic levels of replaceable units are indicated (1) LRU components such as batteries, valves, pumps, etc., and (2) subsystem LRU's such as communications subsystems, engine control units, etc. The selection of the LRU level is a strong driver in the operations segment since it influences flight support facility equipment, warehousing provisions, tools, and orbiter transport provisions. The geometry and size of the units, particularly at the subsystem level, has a significant influence on the SOC storage facilities nd also on the orbiter transportation packaging and handling. The larger subsystem type unit offers advantages for space-basing activities by minimizing the number of required interfaces. Most interfaces

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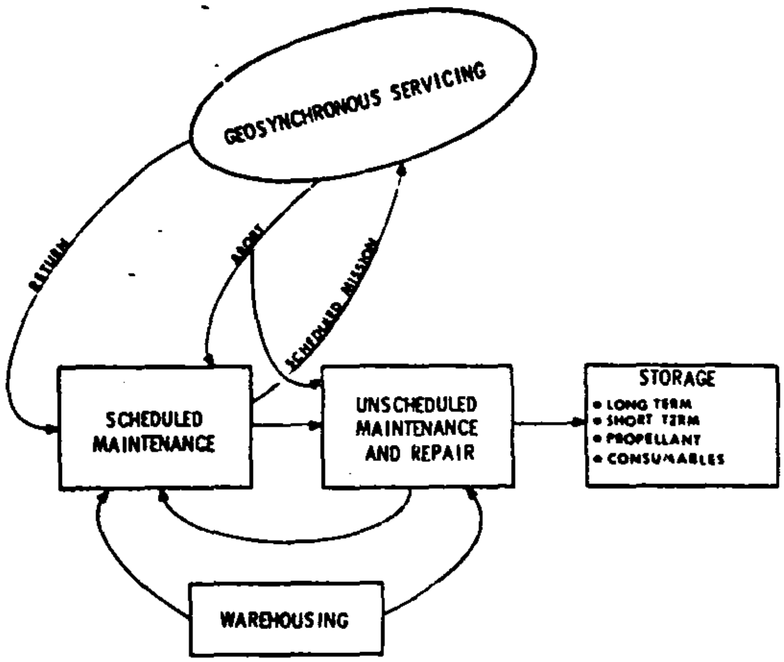

FIGURE 5.1 SOC MAINTENANCE/SERVICING OPERATIONS CYCLE

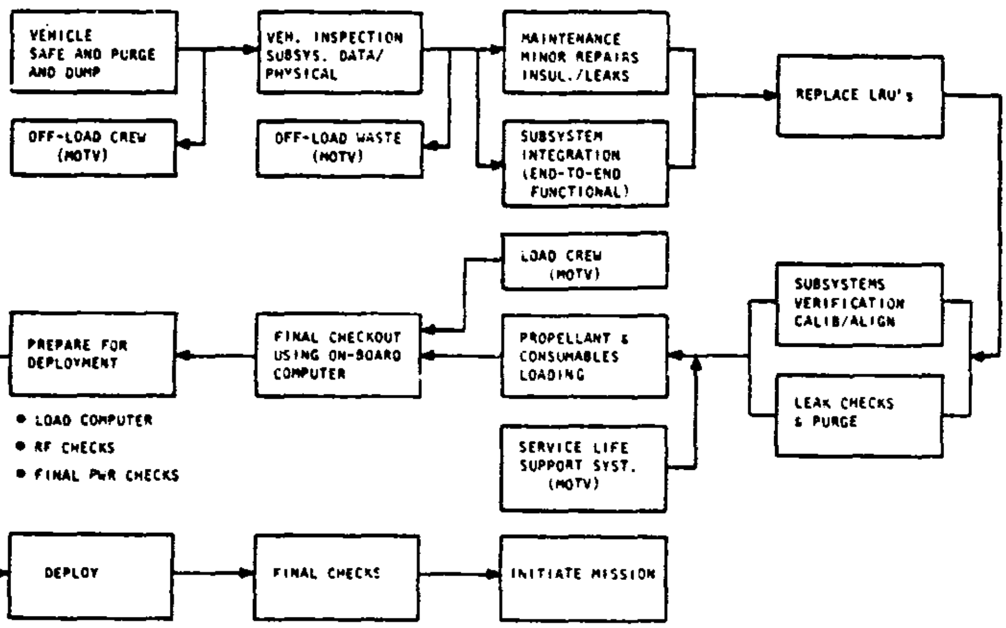

FIGURE 5.2 FLIGHT SUPPORT FACILITY - SCHEDULED FLOW (OTV/MOTV)

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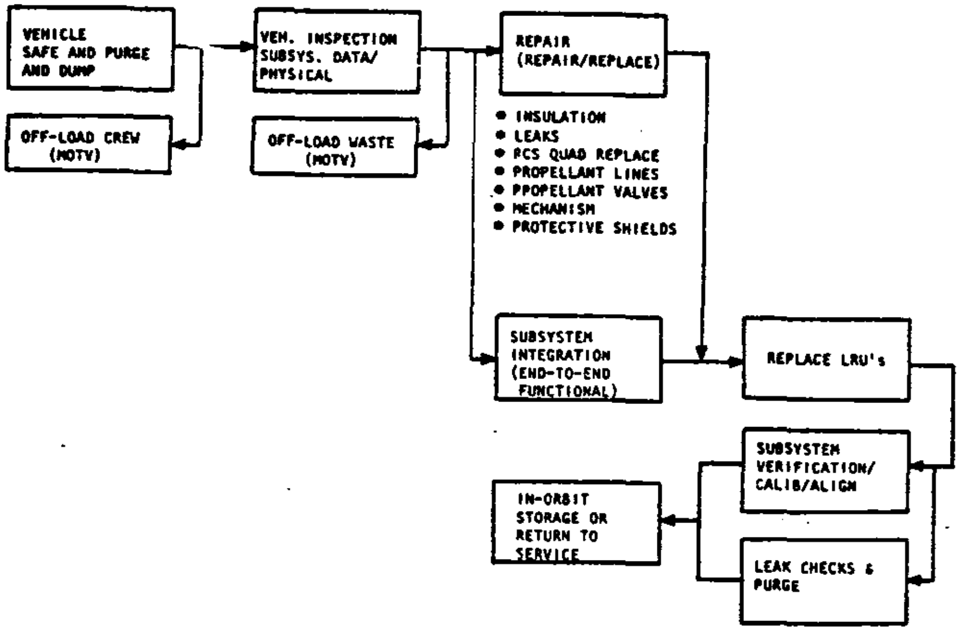

FIGURE 5.3 FLIGHT SUPPORT FACILITY - UNSCHEDULED FLOW (OTV/MOTV)

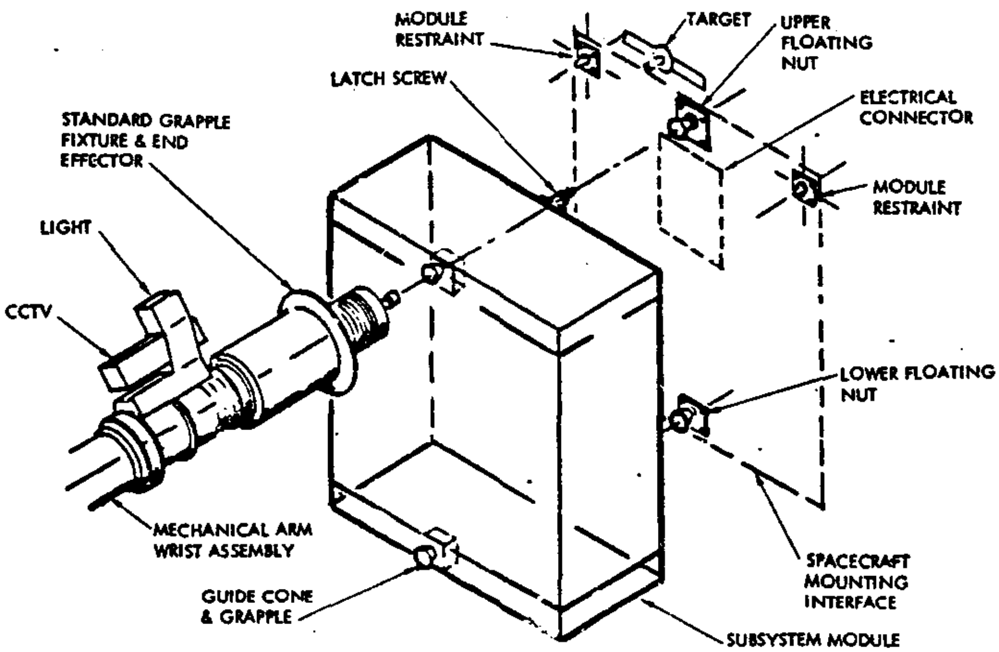

FIGURE 5.4 MULTI-MISSION SPACECRAFT SUBSYSTEM MODULE

will include mechanical fastening and utilities connectors, both electrical and hardline. The larger unit also has the advantage of a more complete entity checkout and verification. An example of this subsystem type LRU is illustrated in Figure 5.4 - The Multi Mission Spacecraft Subsystem Module. As the space operations grow the repair or replacement of units within the subsystem LRU's may become a feasible and desireable activity. This capability at the SOC would enhance the flexibility of the servicing function.

Warehousing

Spare parts for supporting the spacecraft servicing activities are required to be available at the site. The type of items anticipated to be stored at the flight support facility are indicated in Table 5.3. Storage of spacecraft components may well require a conditioned environment. This requirement may be part of the item and need only be supplied with power such as the MMS unit illustrated in Figure 5.4. In other instances, environmental protection may be required. Consequently, an enclosed environment for some items would be desireable. Anticipated long storage periods may also indicate the desireability of meteorite and debris protection which would be afforded by an enclosed facility. Warehousing of large items that may have been identified as unscheduled maintenance items may only require temporary storage preparatory to installation. External storage facilities for such infrequent activities need to be considered. The warehousing function, therefore, requires an enclosed area and an external storage area that may accommodate both subsystem LRU's and temporary storage of large items. The warehouse facility should be sized to accommodate one OTV ship set of spares. The exact nature of these spares need to be determined as part of the space-based OTV concept.

TABLE 5.3 TYPICAL ITEMS IN FLIGHT SUPPORT FACILITY

ITEM	TYPE/FUNCTION WAREHOUSE
LRU's	SUBSYSTEM MODULE BATTERIES
SPECIAL TOOLS AND DEVICES	END EFFECTORS
SPECIAL FIXTURES	SERVICING SUPPORT
REPAIR KITS AND MATERIALS	INSULATION, SEALS, FASTENERS
HANDLING EQUIPMENT	LRU & COMPONENT REPLACEMENT
SPECIALIZED TEST EQUIPMENT	SYSTEMS ANOMALIES CHECKOUT

Storage

This function considers the containment of items such as the MOTV crew module, spare OTV elements as referred to previously, and the storage of propellants and other liquids such as hydrazine, and gases required for preparation of spacecraft flight or for servicing tasks such as purging, i.e., helium. All of these categories require the availability of rather large volumes. The storage of propellants in particular must consider the safety of the SOC. Locating these tanks away from operations activities would be desireable to minimize their vulnerability to damage and consequently be in a safer environment. The MOTV crew module storage location should consider a position that would maintain its habitable environment to minimize reactivation effort. The other large items such as spare OTV elements need to be stored in such a position as not to hinder the normal operations of the SOC, i.e., space construction, spacecraft servicing and orbiter docking.

The storage requirements are a big driver on the arrangement of the total SOC configuration. A critical review of this capability is required in conjunction with the development of the SOC configuration.

5.2.2 EVA and/or Remote Manipulator Operations

A significant issue in SOC servicing operations involves the question of which function should be performed by EVA operations and which should be remotely controlled and involve manipulator operations. During this study such decisions were made informally, on a case-by-case basis as each function in a typical OTV servicing operation was analyzed. However, the general philosophy for such selections developed by Rockwell over a period of five years of construction analysis, is equally applicable to this issue.

Allocations of Servicing Functions to Man and Machine

Fundamental considerations for selecting men or machinery for specific functions and environments are briefly summarized in Table 5.4 (Reference 1). Also, it was observed that space servicing has many critical characteristics in common with space construction, although the degree of importance may be different in several categories. Table 5.5 lists those considered of significance and indicates how they may affect planning and carrying out space servicing/maintenance operations.

A recommended process of methods selection for space construction tasks (including checkout, repair and maintenance) is outlined graphically in Figure 5.5. Note that the potential tasks include assembly and equipment installation which are also performed in space servicing, though typically to a much lesser degree than might be done for constructing a large space platform.

Following identification of viable candidate methods and tasks, the first major methods selection consideration shown is environmental constraints. In the context of low earth orbit SOC operations and current

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TABLE 5.4 MAN/MACHINE CAPABILITIES

Human Superiority	Machine Superiority
1. Originality (ability to arrive at new, different problem solutions)	1. Precise, repetitive operations (Do not suffer physiological fatigue or loss of motivation)*
2. Reprogramming rapidly (as in acquiring new procedures)	2. Reacting with minimum lag (in microseconds, not milliseconds)
3. Recognizing certain types of impending failures quickly (by sensing changes in mechanical and acoustic vibrations)	3. Storing and recalling large amounts of data
4. Detecting signals (as radar scope returns) in high-noise environments	4. Being sensitive to stimuli (Machines sense energy in bands beyond man's sensitivity spectrum)
5. Performing and operating though task-overloaded	5. Monitoring functions (even under stress conditions)
6. Providing a logical description of events (to amplify, clarify, negate other data)	6. Exerting large amounts of force**
7. Reasoning inductively (in diagnosing a general condition from specific symptoms)	7. Reasoning deductively (in identifying a specific item as belonging to a larger class)
8. Handling unexpected occurrences (as in evaluating alternate risks and selecting the optimal alternate, or corrective action)	**Not necessarily true in space construction--especially in the case of the Shuttle remote manipulator system (RMS).
9. Utilizing equipment beyond its limits as necessary (i.e., advantageously using equipment factors of safety)	*The Shuttle RMS accuracy is questionable at this time (as compared to EVA-manual operations)

space suit capabilities, none of these constraints are considered as "show stoppers." However, it is generally undesireable to expose the crew to the hazards, discomforts and fatigue of extravehicular operations unless there are strong advantages in doing so.

The next major constraints of concern are classified as involving location constraints. Experience has shown that a difficult minimal clearance access problem may be created by specific spacecraft configurations, by servicing equipment or by a combination of both. EVA operations may be a preferred means to solve such access/reach problems. Another possibility is that any selected manipulator device may be unable to reach far enough to accomplish a specific task. Here a crew member, using

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TABLE 5.5 SIGNIFICANT CONSIDERATIONS FOR ALLOCATION OF SERVICING FUNCTIONS

GENERAL SPACE OPERATIONS WORK CONSIDERATIONS	SPECIFIC SERVICING METHODS CONCERNS
• TIME CONSTRAINTS (PREPARATION, REST, RECHARGE, ETC.)	• CREW PRODUCTIVITY, ON-ORBIT STAYTIME COSTS
• RATES OF WORK ACCOMPLISHMENT	• SELECTION OF TRANSPORT METHODS, TOOLS, DEVICES
• ACCURACY	• SELECTION OF SERVICING EQUIPMENT, CREW, TOOLS
• DEXTERITY	• DECISIONS ON MEN VS. MACHINE
• SIZE AND REACH, TRAVEL DISTANCE	• SIZING MANIPULATORS, SELECTING METHODS
• MASS PROPERTIES (WEIGHT, MOMENTS OF INERTIA LIMITS)	• USE OF MACHINERY VS. CREW FOR TRANSPORT
• VISION AND LIGHTING	• PLANNING LOCATION, ACCESS, EQUIPMENT USE, POWER
• SYSTEM INTERFACE REQUIREMENTS	• DESIGN FOR COMPATIBILITY, MINIMUM IMPACT
• POWER REQUIREMENTS	• POWER SOURCES, ENERGY SUPPLY REQUIREMENTS, WIRING
• RELIABILITY	• TYPE, NUMBER OF EQUIPMENT OR CREW
• SAFETY AND CONTINGENCIES	• DECISIONS ON DIRECT VS. REMOTE CONTROL
• ENVIRONMENTAL CONSTRAINTS	• SELECTION OF METHODS, CREW PROTECTION
• SPECIAL HUMAN FACTORS AND MISSION CONSIDERATIONS	• CREW ACCEPTABILITY, SKILLS MIX OF CREW
• COST CONSIDERATIONS	• COMPARISONS OF METHODS FOR SERVICING

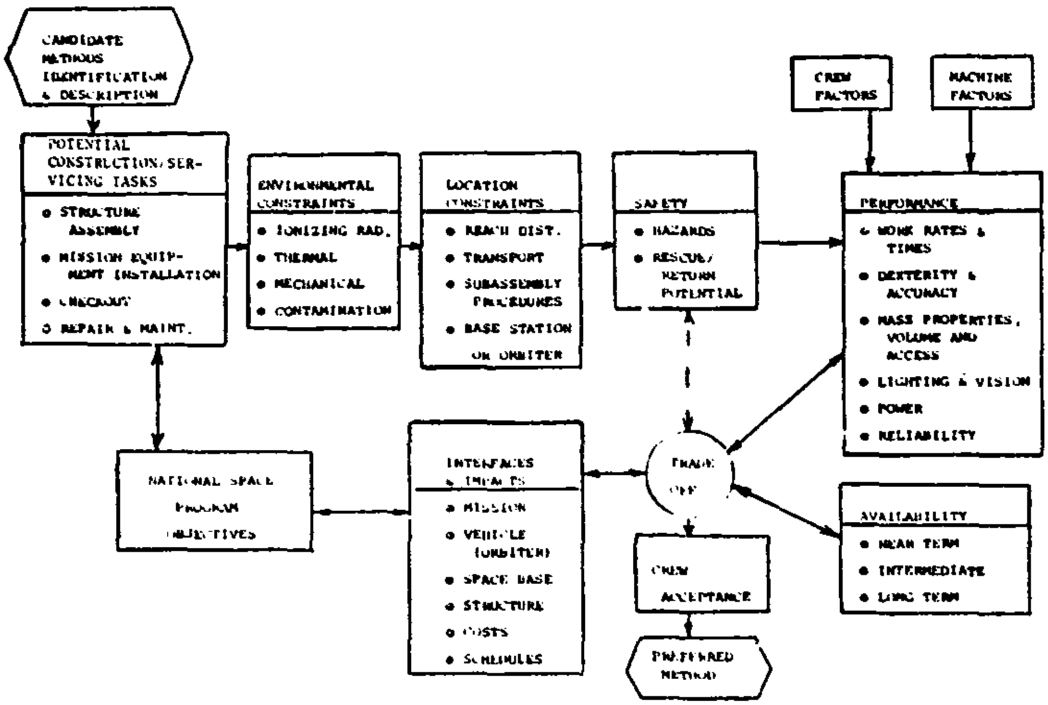

FIGURE 5.5 RECOMMENDED METHODS SELECTION PROCESS FOR LARGE SPACE STRUCTURES CONSTRUCTION

either hand rails or a manned maneuvering unit (MMU), could potentially solve the problem in an economical manner. However, the basic approach in this study was to select (or design) versatile equipment which could minimize problems of access and reach.

Another major consideration is safety of the SOC crew, in the context of the remaining potential hazards involved in the foregoing approaches. Of particular interest to space servicing are such concerns as fuel transfer, snags, impacts from moving equipment, electrical shock, and stored energy.

The next set of considerations strongly involves performance considerations and productivity. It also involves trades of development costs and schedules versus availability of equipment, potential impact on involved systems (SOC, serviced vehicle, orbiter and support equipment) and crew acceptance. Here the general considerations in Table 5.4 are translated into quantitative and qualitative comparisons of specific technical solutions, such as crew dexterity and RMS accuracy, transport velocity versus mass and moment of inertia, power and energy demand versus supply, and estimated reliability. In general, one value of the concept of a SOC is presumed to be minimization of impacts on the basic Shuttle orbiter design. Such considerations tend to favor increased specialization of SOC equipment and development of a relatively larger, more powerful and generally capable remote control capability in its crew stations. However, it is also presumed possible and desirable to design EVA equipment, select SOC cabin pressure levels and design airlocks, etc., to enhance EVA operations in the vicinity of the SOC.

The consequences of this approach are that no EVA would be planned from the orbiter docked or berthed at the SOC, and favor would be given to those remote handling functions normally performable by standard RMS configurations. However, a docking module and a handling and positioning aid were considered as potential new equipment items.

Table 5.6 summarizes some general guidelines adopted for this study, their supporting rationale and related specific examples of implementation methods are described in the Servicing Activities Data Sheets, Appendix D.

5.2.3 Facility Growth

An operations philosophy was developed that was concerned with describing the early flight support facility operations and determining how to grow the facility to its full up capability.

Initially, the flight support facility operations may involve only the support of a single stage OTV performing a LEO to GEO and return type mission. This type of mission can be accomplished with a relatively modest flight support facility. Figure 5.6 illustrates the type of missions anticipated. Refueling of the OTV stage is anticipated to be accomplished directly from the orbiter.

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As the flight activity increases it may be desirable to add fuel storage capability to the SOC. Continued activity increases will probably include MOTV missions. A two stage propulsion configuration appears likely for an MOTV mission. At this stage of increased mission activity a flight support facility growth to include the capability to service MOTV's is necessary.

TABLE 5.6 GUIDELINES AND IMPLEMENTATION SAMPLES RELATING TO MAN/MACHINE FUNCTION ALLOCATION IN SERVICING

GUIDELINES	RATIONALE	SOC-SERVICE FIXTURE IMPLEMENTATION
TRANSPORT • TRANSPORT HIGH-MASS, BULKY OBJECTS BY REMOTE MANIPULATOR RATHER THAN EVA. EXCEPTION: CLEARANCES COMPLEX AND CRITICAL.	• HIGHER VELOCITY TYPICAL WITH REMOTE MANIPULATOR. • TV SCREEN HAS MINIMAL DEPTH PERCEPTION. STEREO TV IS COMPLEX.	• TROLLEY-MOUNTED REMOTELY CONTROLLED MANIPULATOR ON SERVICE FIXTURE (SF). • RMS & PRESSURIZED CABIN ON R/CM • ORBITER RMS FOR EXTRACTION FROM CARGO BAY
• IF LARGE MODULE TRANSPORT CLEARANCES ARE COMPLEX & CRITICAL, USE EVA/CHERRY PICKER OR EQUIVALENT.	• CREW DEPTH PERCEPTION • CREW WIDE RANGE OF INSTANTANEOUS VISION • ON-SITE OBSERVATION DURING TRANSPORT.	• CHERRY PICKER ON TROLLEY-MOUNTED REMOTE MANIPULATOR FOR SERVICE FIXTURE OPERATIONS
WORK STATION OPERATIONS • USE EVA FOR HIGHLY DEXTEROUS OPERATIONS OF LOW FREQUENCY. • USE EVA FOR CLOSE INSPECTION REQUIRING JUDGMENT OF QUALITATIVE NATURE.	• DEMONSTRATED CAPABILITY AND LOW DEVELOPMENT COST (NO COMPLEX MACHINERY FOR CRITICAL TASK).	• EVA/CHERRY PICKER ON TROLLEY-MOUNTED REMOTE MANIPULATOR FOR GENERAL INSPECTION AND CONTINGENCY REPAIR/REPLACEMENT OPERATIONS
• USE REMOTE MANIPULATORS FOR REMOVAL/REPLACEMENT OF STANDARD LINE REPLACEABLE UNITS DESIGNED FOR ROUTINE SERVICING.	• MINIMIZE EVA EXPOSURE TO DISCOMFORT, HAZARDS, AND DON/DOFF TIME LOSS. • TAKE ADVANTAGE OF DESIGNS FOR PREDICTABLE HANDLING OPERATIONS.	• USE OF SERVICE FIXTURE MOUNTED REMOTELY OPERATED MANIPULATORS FOR OTV SUBSYSTEM LRU's DESIGNED FOR REMOVAL/REPLACEMENT BY MANIPULATOR

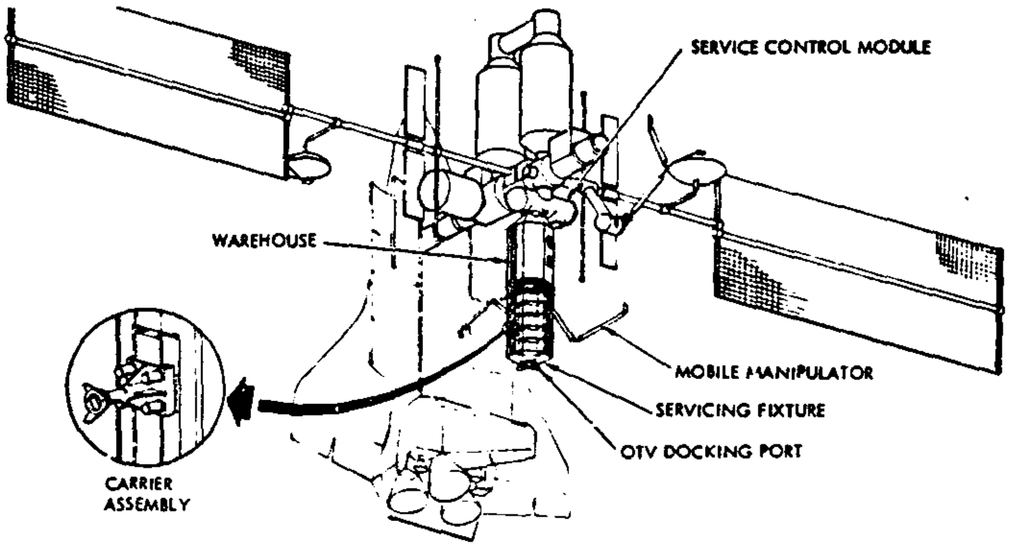

FIGURE 5.6 INITIAL FLIGHT SUPPORT FACILITY CONFIGURATION

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The flight support facility developed for this task illustrates the initial SOC operations arrangement and the growth arrangements in Drawing 42690-019, Appendix A.

5.3 FLIGHT SUPPORT FACILITY DEVELOPMENT

The flight support facility occupies the -Z quadrant of the referenced SOC configuration with the stage assembly module (SAM) attached to the service module as indicated in Figure 3.1. As configured, however, a potential interference exists between the SAM and the vertical fin of the orbiter when the docking 6° misalignment tolerance is imposed. Various arrangement options, Figure 5.7 were investigated that would meet the general requirements for orbiter clearances, MOTV crew egress, and non violation of the space construction sector. The selected arrangement, option 2 of Figure 5.7, met the general requirements, but contributed to increased drag area and presented a potential Z axis assymetric arrangement. A structural adapter to support the SAM and MOTV was necessary to accomplish this arrangement. However, further development of this arrangement has indicated the desirability of this concept which is highly flexible in configuration, accommodations and support to OTV operations.

The structural adapter not only provides the support to the SAM but also provides the pressurized interface between the crew module of an MOTV and the SOC. This capability, therefore, permits the location of a servicing control center to be incorporated within the adapter.

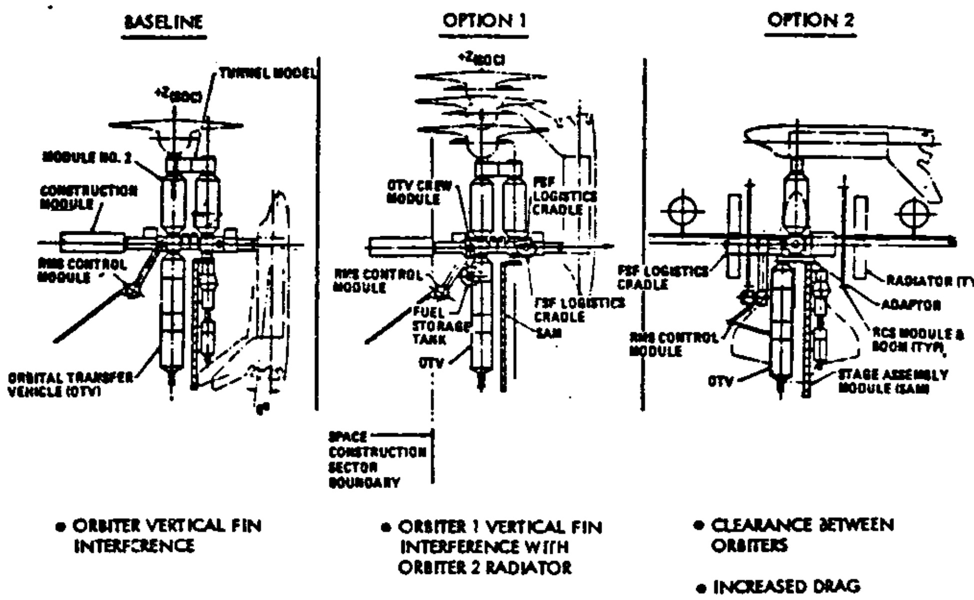

FIGURE 5.7 CONFIGURATION OPTIONS

The selected flight support facility which consists of the service control module, formerly referred to as the structural adapter, and a servicing fixture, formerly referred to as the SAM is now located in the SOC pressure Volume 2, Figure 5.8. The service control module is envisioned to contain all of these controls and displays required to support the flight support facility operations and, in addition, can contain a redundant set of basic SOC station operations controls and displays. This arrangement, therefore, can provide the SOC emergency control back-up station. This back-up control capability concept dictates that the prime SOC control center be located within the SOC pressure Volume 1.

5.4 SERVICING FIXTURE CONCEPT

One of the major operational facilities that characterize the SOC is a space vehicle support facility. The facility is required to provide a berthing capability to a variety of space vehicles and to assemble, inspect, maintain and service the space vehicles. The objective of this task was to determine a servicing fixture (SF) arrangement that provides all the required servicing facilities for a Manned Orbital Transfer Vehicle (MOTV) as representative of the types of space vehicles that the SOC is required to accommodate. To define a feasible arrangement, a set of criteria by which to guide the development of the arrangement was established as shown in Table 5.7. Similarly, the characteristics of a reference MOTV concept were derived jointly by Rockwell International/Boeing Aerospace/NASA/JSC. The derived MOTV characteristics are:

(1) Assume the crew module is a separate module which can be detached from the propulsion stages of the OTV.

(2) Further assume the overall OTV configuration contains an avionics module which can operate with or without a crew module. It will provide the command and control functions for the OTV when operated as an unmanned transfer stage but would also interface with the crew module for on-board control functions when flown as an MOTV.

(3) It is currently envisioned that unmanned OTV utilizations will outnumber manned OTV missions by about three to one.

(4) Thus, a SOC requirement exists for stowing the crew module until it is needed.

(5) Consider the MOTV crew complement to be two persons.

(6) Assume the MOTV (crew module) does not have an airlock. EVA during MOTV missions will be provided by cabin depressurization.

(7) Pressurized access between the MOTV crew module and the SOC must be provided during normal crew exchange and servicing operations. It may also be considered for use during periods of crew module stowage to provide extra habitability volume when both the SOC crew and the MOTV crew are present.

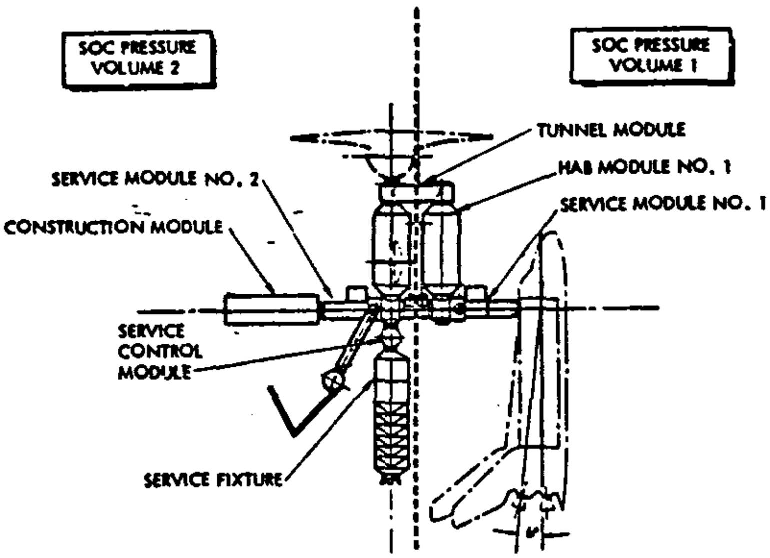

FIGURE 5.8 SOC PRESSURE VOLUMES

TABLE 5.7 SERVICING FIXTURE ARRANGEMENT CRITERIA

- ACCOMMODATE BOTH TANDEM AND PARALLEL TANK/STAGING OTV CONCEPTS
- MINIMIZE CONFIGURATION-INDUCED FORCES ON SOC CONTROL (DRAG, ASSYMETRY, ETC.)
- PROVIDE SERVICING FACILITIES FOR OTVs, PLANETARY VEHICLES, AND SATELLITES
- PROVIDE FUEL STORAGE FACILITY
- PROVIDE SERVICING CONTROL CENTER
- MAINTAIN ORBITER CLEARANCE FOR BOTH A ONE-ORBITER ARRANGEMENT AND A TWO-ORBITER ARRANGEMENT
- PROVIDE RCM ACCESS AND VISIBILITY TO SERVICING VEHICLES
- PROVIDE DEDICATED PORTS FOR:
  - OTV CREW MODULE
  - LOGISTICS MODULE/CRADLE
  - MOTV

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(8) It is envisioned that the MOTV/OTV turnaround operations will include an EVA "walkaround" inspection. This could involve the use of one EVA crewman plus one EVA crewman as backup. Crew workload and task assignments should consider this function in conjunction with SOC manning levels down to as few as four crewmen. The role of the MOTV crew in these inspection operations should also be considered.

(9) Although the main emphasis is for space turnaround of the MOTV/OTV vehicles, the occasional return to earth (after eight flights) for major ground overhaul should be included in the determination of space servicing functions.

(10) It is envisioned that all OTV/MOTV elements will have self-diagnostic capability with built-in computer switching to redundant units. This interacts with the amount of pre-knowledge about failed equipment and servicing needs which will be available before the OTV returns to SOC for servicing.

(11) Tandem and parallel tanking/staging arrangements should be considered to determine the design implications of the flight support facility.

(12) The implications of both aero-braking and propulsive braking MOTV concepts on SOC based servicing and turnaround operations should be considered.

(13) Assume that SOC will provide sufficient support provisions for the MOTV crew so as not to require a special Shuttle flight for returning the crew to earth.

The MOTV characteristics were the basis for a MOTV turnaround flow analysis the result of which is shown in Figure 5.9. The flow chart was developed in conjunction with two MOTV configurations, a three-drop tank arrangement developed by Grumman Aerospace Corporation (Reference 2) and a tandem tank arrangement as shown in Figures 5.10 and 5.11 respectively.

The major activities shown in Figure 5.9 were analyzed individually to determine what features must be included within the SF so it can perform these activities. The individual servicing activity data sheets which resulted from the analysis are contained in Appendix D. The aggregate result of all the servicing activity data sheets is the SF arrangement shown in Figures 5.6 and 5.12. The major components of the SF are a control module and a hexagonally shaped structural strongback that supports several subsystems such as berthing/docking ports, two manipulators, refueling provisions and storage compartments.

5.4.1 Structural Configuration

The SF structure is a linear shaped truss strongback with a hexagonal cross section approximately 15.2m (50.0 ft) in length as shown in Figure 5.13. The hexagonal cross section measures 4.45m (175.00 inches)

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across the diagonal which is within the payload envelope of the orbiter cargo bay. Similar considerations, including allowances for the docking module, dictated the length of each section.

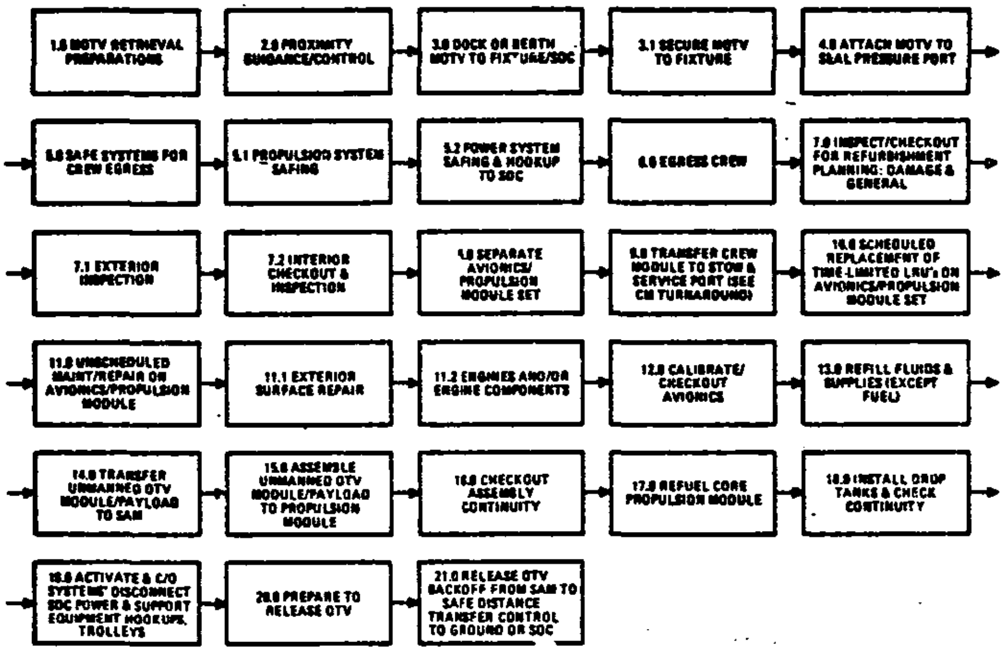

FIGURE 5.9 MOTV TURNAROUND FLOW CHART

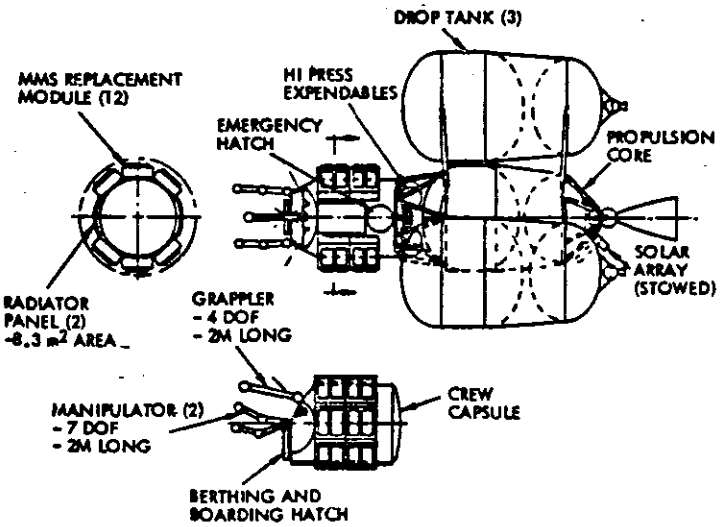

FIGURE 5.10 3-DROP TANKS MOTV CONFIGURATION

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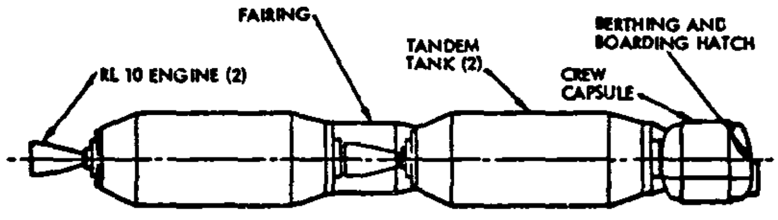

FIGURE 5.11 TANDEM TANKS MOTV CONFIGURATION

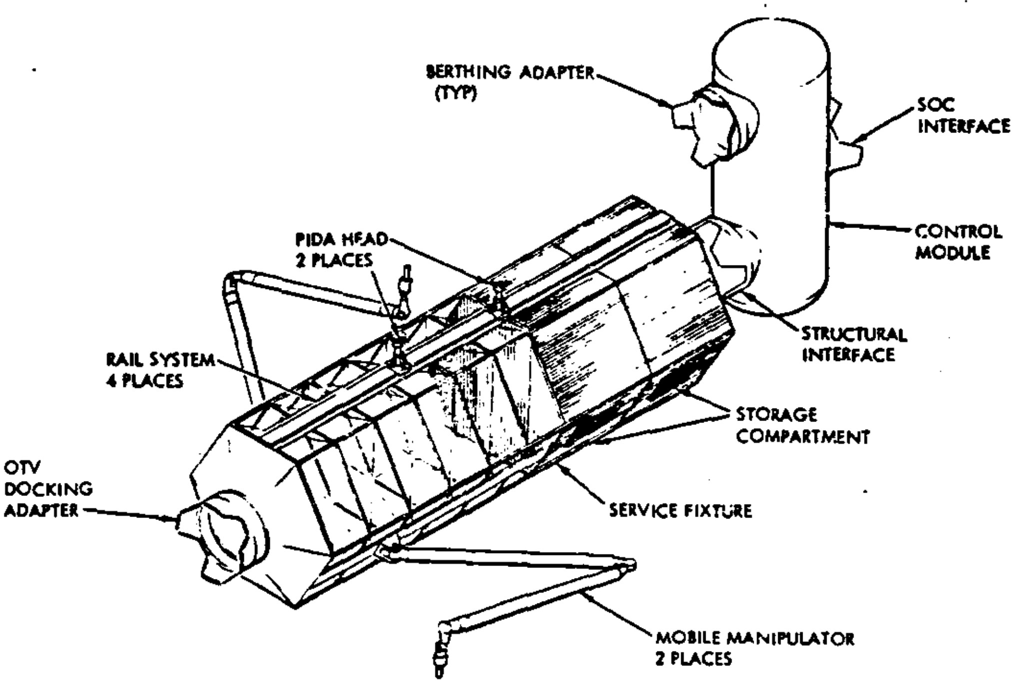

FIGURE 5.12 SERVICE FIXTURE ARRANGEMENT

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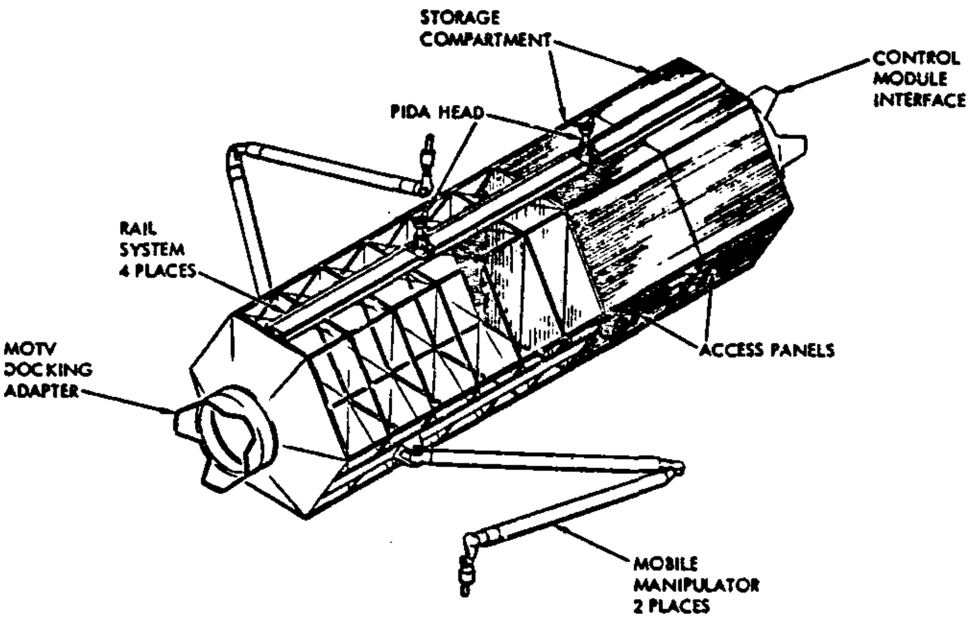

FIGURE 5.13 SERVICE FIXTURE STRUCTURE

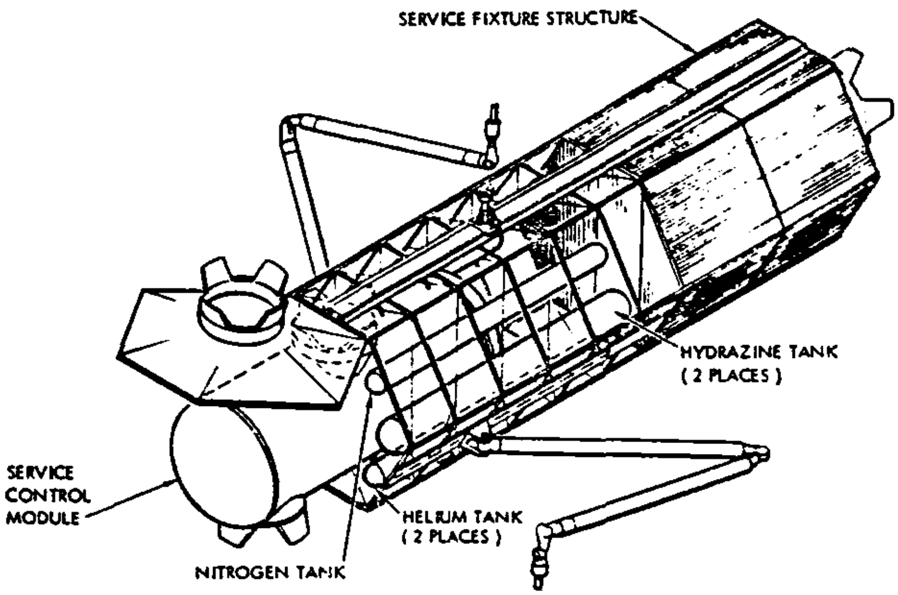

FIGURE 5.14 SERVICE FIXTURE PACKAGING CONCEPT

The SF structural assembly is transported to the SOC as a fixed structure. The control module is sized to nest within the SF so that the total initial flight support facility can be delivered to the SOC with one shuttle launch as shown in Figure 5.14. The SF features two berthing/docking ports, one on each end, and translation rail systems on four of the six hex faces. Two storage compartments and two mobile manipulators are supported by two of the translation rail systems. One port of the first section interfaces with the control module. Its second port is utilized for the docking of returning space vehicles.

5.4.2 Control Module

To operate the entire flight support facility complex, a control module is provided as shown in Figures 5.12 and 5.15. Configured as a pressure vessel and arranged to be an integrated part of the SOC, the control module offers two important provisions. By allowing all SF control functions to be incorporated in its design, the control module frees the SOC proper from that activity. Secondly, it provides a stowage port for the MOTV crew module during unmanned OTV missions.

The control module is a cylindrical shaped pressure vessel which features three SOC-type berthing ports, only two of which are needed for manned access. One port is located centrally on the cylindrical section and is utilized for mating with the SOC. The other two ports are also located on the cylindrical section 180° away from the first port. One of

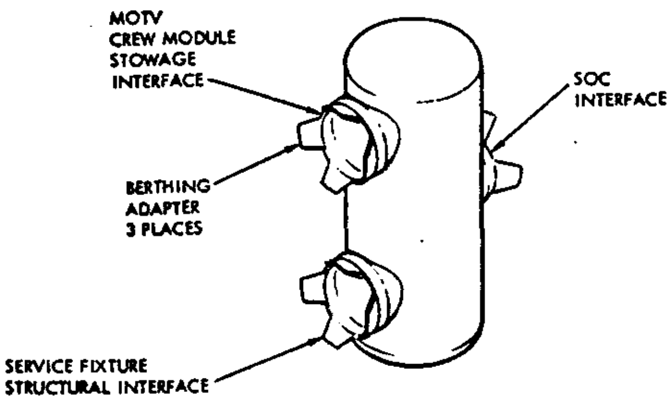

FIGURE 5.15 CONTROL MODULE

these two ports does not require manned access and it is utilized strictly as a structural attachment for the SF strongback. The third port functions as a stowage position for the MOTV crew module and as an egress path of its crew into the SOC. When the MOTV crew module, the SF control module and SOC are mated together, an integrated shirt-sleeve environment is provided among all the pressurized modules.

5.4.3 Servicing Fixture Subsystem Provisions

The SF must incorporate several major subsystem provisions so it can fulfill its function as a complete servicing facility. Most significant of these provisions are the berthing/docking ports, translation rail systems, mobile manipulators, fluid transfer provisions and spare parts stowage compartments. These provisions which are depicted in Figure 5.12 are described separately along with some of their operational requirements which influenced their configurations.

Berthing/Docking Ports

The structural strongback features two berthing/docking ports, one on each end. These ports can mate with any of the SOC ports and possess several commonality features with them such as latches and guides. One port which will be utilized for the docking of returning OTVs and other space vehicles must incorporate the shock attenuation capability. Whereas those used strictly as structural attachments and normally mated by berthing operations will be passive. With that guideline, one passive and one active port, i.e., with shock attenuation provisions and utility interfacing mechanisms, will be required on the structural segment. It should be noted that unlike all other SOC ports, these ports need not provide for manned access. Consequently, all provisions necessary for manned accessibility such as seals and life sustaining utilities will not be required.

Translation Rail System

Four separately operated translation rail systems are needed to support the SF operational activities as seen in Figure 5.16. Two of the systems are at opposite faces of the hexagonal SF crosssection (180° apart) and are intended to support and position OTVs and/or other space vehicles at the proper SF work station during assembly and servicing operations. The other two systems support and position two mobile manipulators which are utilized as assembly and servicing tools.

Two of the rail systems consist of a support rail that covers the total length of the SF, and a transporter unit that can travel the length of the support rail. The transporter contains a PIDA head device (Figure 5.17) that interfaces and supports the space vehicle being serviced.

The remaining two rail systems support the mobile manipulators. The manipulators and their typical usage are illustrated in Figure 5.16. The manipulator is a remotely controlled device very similar to the RMS. It

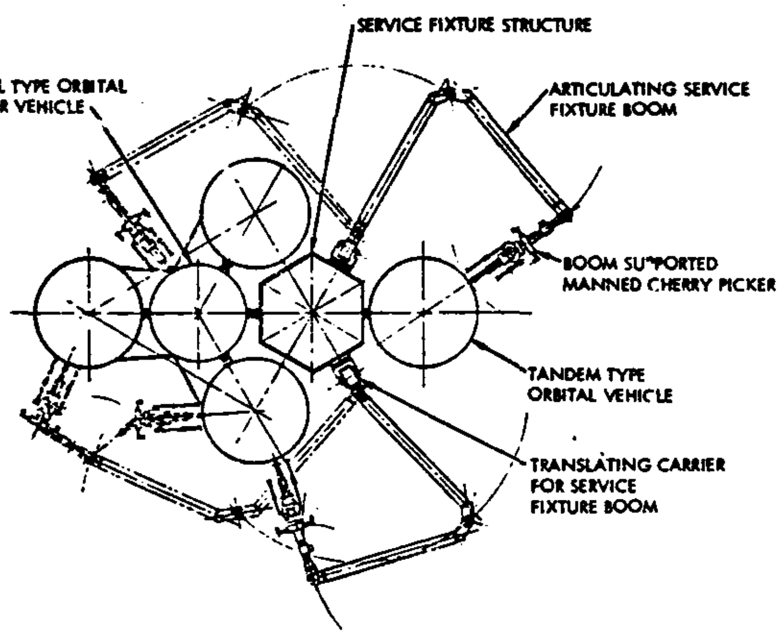

FIGURE 5.16 TRANSLATION RAIL SYSTEM AND HANDLING BOOM REACH CAPABILITY

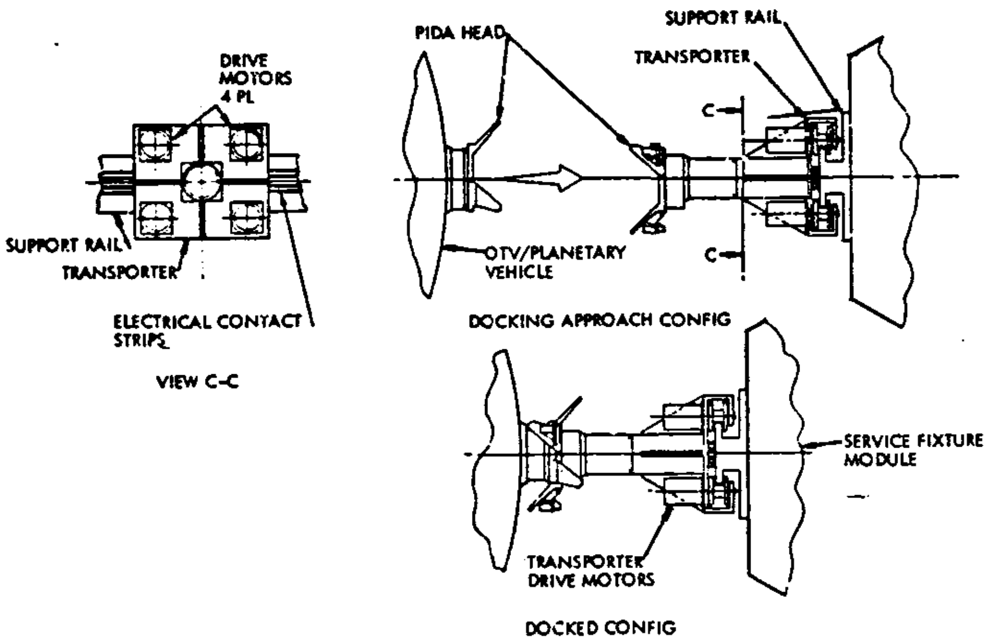

FIGURE 5.17 TRANSLATION RAIL SYSTEM WITH PIDA SUPPORTS

consists of three arms, two equal lengths of 6.1M (20.00 ft) each and a wrist of 1.78m (70.00 in) length. The wrist will include an end effector for interfacing with the various elements of the space vehicle. Each of the manipulators is envisioned to have six degrees of freedom (similar to the RMS) in addition to its mobility along the translation rail system. The elbow joint is offset from the center line of the upper and lower arms sufficiently to allow the manipulator to be stowed in a folded position.

Fluid Transfer Provisions

An important activity of any manned space base such as SOC is the capability to refuel space vehicles in space. The contributions of the SF to this capability include the routing of transfer lines along the SF, the capability to make the connections of the fuel line to the space vehicle, and to provide the pumping power to complete the refueling operation. The SF arrangement of Figure 5.12 provides for routing of fluid lines, valves, pumps, etc., internal to the SF structure. Each fuel line ends with a single degree of freedom arm and a quick connect/disconnect device. Once the OTV/space vehicle is positioned at the refueling station, the swing arm can be activated and the refueling operation performed as suggested by Figure 5.18.

Storage Provisions

The configuration of the servicing fixture features a large internal volume that is dedicated for the storage of LRU's, other spare parts, tools, and special equipment. Figure 5.12 indicates this area on the SF and Figure 5.19 illustrates the storage arrangement concept. Seven large (1Mx1Mx.5M) subsystem modulator LRU's can be accommodated on each of the compartment bulkheads. A swing arm handling device that has the capability to transport and exchange these large LRU's and other types of articles stored in this compartment is also provided. The compartment can be accessed by either one of two doors on the faces of the hexagonal shaped SF structure. When opened, the doors fold and retract inside the SF structure in a manner that will not interfere with the LRUs handling and exchanging operations. The arm has five degrees of freedom and can handle/exchange LRUs on both bulkheads of the storage department.

External storage facilities are also available if desired. Figure 5.19 indicates this capability illustrating the storage of the large LRU's on the external surface of the SF.

On orbit propellant storage provisions are located on the +Y side of the SOC between the solar array support boom and the space construction sector boundary. Two identical tanks are supported from a strong back structure that is attached to the berthing/docking port at that location. The tanks are sized so that radiator heat rejection interference is minimized. Additional tanks can be accommodated from the strong back when required for additional growth capability. The installation of this capability on the SOC can be made at any stage of the evolutionary growth capability so desired.

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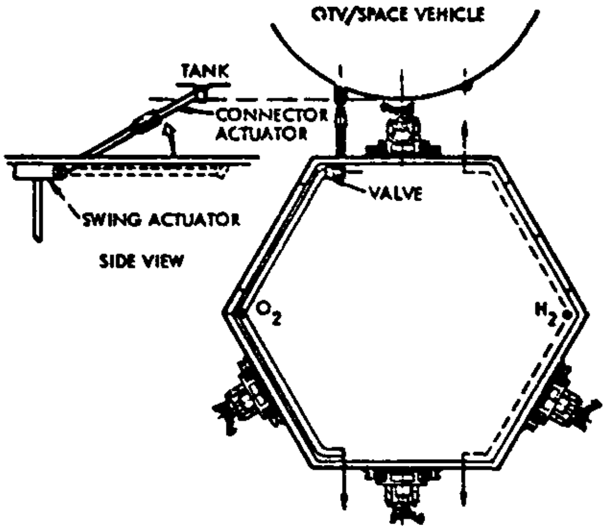

FIGURE 5.18 OTV REFUELING CONCEPT

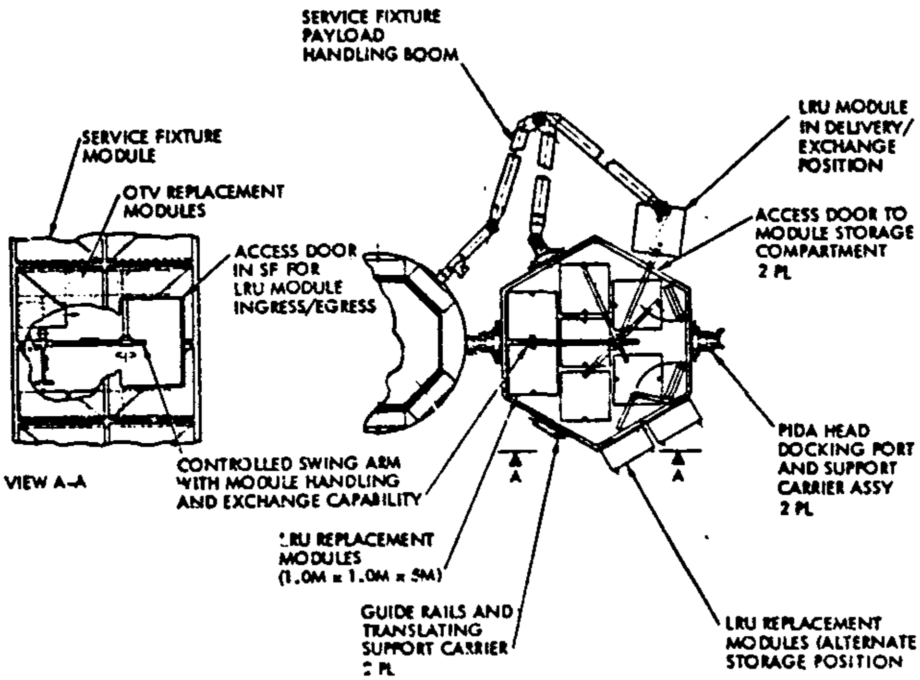

FIGURE 5.19 SERVICE FIXTURE STOWAGE PROVISIONS

5.4.4 Growth Capability

The growth capability of the SF is accomplished by installing another section of the initial SF structure as shown in Figures 5.20 and 5.21. The growth section is berthed to the initial SF section at the end port that was formerly used as the docking port for returning OTV's. The growth increment now contains the OTV docking port. Two additional rail systems are contained on this growth section. The additional rails provide increased capability to simultaneously service various space vehicles. All of the rail systems are extended across the interface between the initial and growth service fixtures. The two additional rail systems are also installed on the initial section, thus providing the capability to service vehicles on all six faces of the flight support servicing fixture. The growth configuration will have sufficient length and the capability to support and service a tandem, 2 stage, MOTV, as shown on Drawing 42690-019.

5.4.5 OTV Implications

The implications or requirements imposed on the OTV/MOTV as determined from the development of the servicing functions have been identified in the previous discussions. Each of the significant implications will be described in this section.

The principal implication is that the OTV stage be initially designed for space based operations including refueling, servicing and maintenance, and some level of repair activities. This implies that the structure as well as the subsystems servicing concepts, engine installations, insulation installation, all be designed specifically for space operations. The shuttle launch criteria, however, is still applicable and must be considered. However, auxilliary devices that could minimize the imposed criteria on the vehicle and would be considered as shuttle flight support equipment should be seriously considered. The ultimate design will then be the minimum weight OTV for space based operations.

The servicing functions trades and analysis indicated the desirability to provide subsystem modular units that are mounted externally at the electronics and systems section of the OTV. This arrangement permits access to these units (LRU's) by the mobile manipulators of the servicing fixture. The arrangement also allows for EVA operations if required.

Statusing of the OTV systems prior to servicing and the check-out operations are performed by an umbilical(s) that interconnects between the facility servicing control center located in the service control module and the OTV. This arrangement permits monitoring of the systems status by the crew in a shirt sleeve environment.

Other umbilical interfaces have been identified for the propellant refueling operations, and the refilling of the other fluids and gases necessary for the operation of the OTV and MOTV stages. Interface provisions for line and tank purging operations have also been identified.

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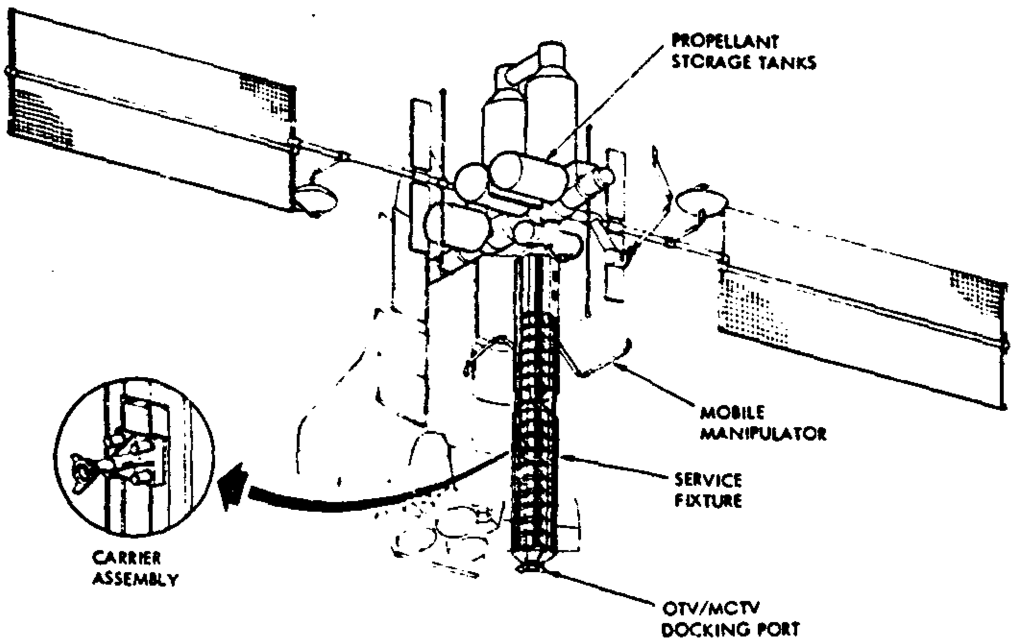

FIGURE 5.20 FLIGHT SUPPORT FACILITY GROWTH CONFIGURATION

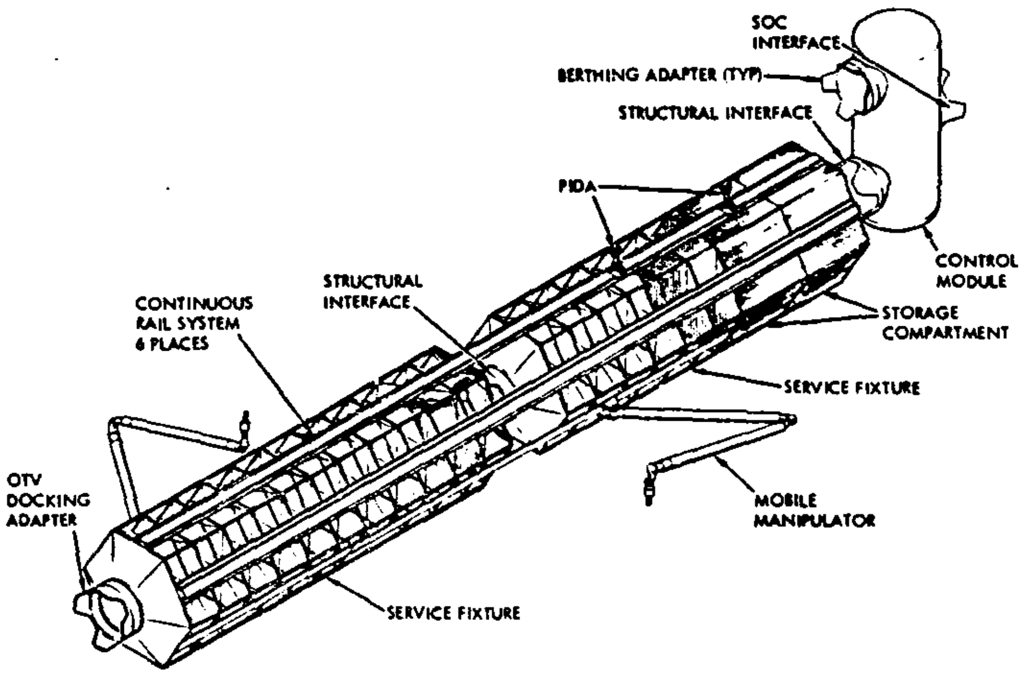

FIGURE 5.21 SERVICE FIXTURE GROWTH CONFIGURATION

An independent crew module that can be attached or detached and stored on the SOC was a requirement identified in the jointly developed MOTV/OTV model. A berthing port and utilities interfaces to accommodate this requirement is therefore, imposed on the OTV. This berthing/docking port also provides the initial attaching provisions for the OTV when returning to the SOC for servicing. The port is centrally located at the forward end of the vehicle.

The provisions on the OTV for the attachment to the servicing fixture utilizes the PIDA provisions that have been identified for the controlled deployment of the OTV stage from the orbiter's payload bay. No additional provisions, therefore, are required to facility the securing of the OTV to the servicing fixture. Grappling provisions are required, however, for the transportation of the OTV stage from the orbiter to the servicing fixture. The same grappling fixture may be utilized to transport the docked, returned OTV from the end of the fixture to the fixture translation rail system.

Additional capabilities that would permit the replacement of large items such as the main engines and insulation panels need to be determined by performing trades of space based replacement vs ground replacement.

5.5 MOTV/OTV OPERATIONS FROM THE SHUTTLE

The objective of this task is to determine the requirements/implication on the Shuttle and on the MOTV to perform flight support services from the Shuttle as an evolutionary step to the implementation of the SOC.

The analysis concerning servicing of an MOTV/OTV from the SOC has identified certain requirements that are imposed on the OTV. This task that is concerned with the assembly, launch, and servicing of an OTV from the Shuttle Orbiter uses an OTV model that has incorporated the requirements as defined from the SOC servicing analysis. This OTV model, therefore, represents a space-based design. Table 5.8 lists the significant provisions that are incorporated on the space-based design OTV that are associated with the servicing and handling operations.

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TABLE 5.8 SPACE-BASED OTV SERVICING PROVISIONS

• STANDARD BERTHING PORT AT FORWARD END
• TWO PIDA ATTACH PROVISIONS ON BODY
• GRAPPLE FITTINGS TO ACCOMMODATE RMS
• EXTERNALLY MOUNTED SUBSYSTEM PACKAGES (LRU)
• ELECTRICAL UMBILICAL INTERFACE
• PROPELLANT FILL INTERFACE
• HYDRAZINE & HELIUM FILL INTERFACES

OTV Orbit Assembly Operations With the Orbiter

OTV in-orbit assembly operations utilizing the orbiter in the conduct of LEO to GEO missions is a precursor to MOTV/OTV operations from the SOC. In-orbit assembly of the tandem configuration MOTV/OTV requires three support flights of the orbiter to stage a launch mission. For this study the LEO to GEO mission is assumed to be a payload delivery mission.

The three orbiter flights supporting OTV assembly are as follows:

(1) Delivery of propulsion unit to orbit

(2) Delivery of payload to orbit and assembly with propulsion unit

(3) Delivery of propellant to orbit, fuel transfer, and final checkout

The first flight delivers the OTV to orbit. The PIDA provisions on the OTV permit the deployment of the OTV from the payload bay. Checkout of the OTV from this position appears feasible. Checkout would be accomplished with the aid of an umbilical line interfacing with the provisions on the OTV. The OTV would be deployed from the PIDA holding position by the RMS after which the OTV stabilization system is activated.

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The second flight consists of the payload package and a docking module adapter. The adapter is a unit that interfaces with the standard docking/berthing interface and creates an interface that will accept the PIDA provisions on the OTV, Figure 5.22. After the adapter installation has been made, the RMS captures the OTV and places it on the docking adapter, Figure 5.23. This arrangement holds the OTV in position to accept the payload. The payload is transported by the RMS and is attached to the OTV utilizing the standard berthing port. This assembly is then placed into orbit after undergoing a checkout procedure and again activating the stabilization system.

The third flight fuels the assembly through the appropriate interfaces while it is secured to the docking adapter. Leak checks are performed after refueling. After final checkout the assembly is deployed from the orbiter by the RMS and the OTV mission commences.

Servicing of the OTV is accomplished from its position atop the docking/berthing adapter. This position permits the RMS itself or with the open cherry picker (OCP) to perform the services necessary to prepare the OTV for another mission.

No warehousing capability is provided for this mode of OTV servicing. Consequently, the degree of on orbit servicing from the orbiter will depend on the condition of the OTV, the availability of orbiters for transport of spare parts, and ultimately the decision of the mission director.

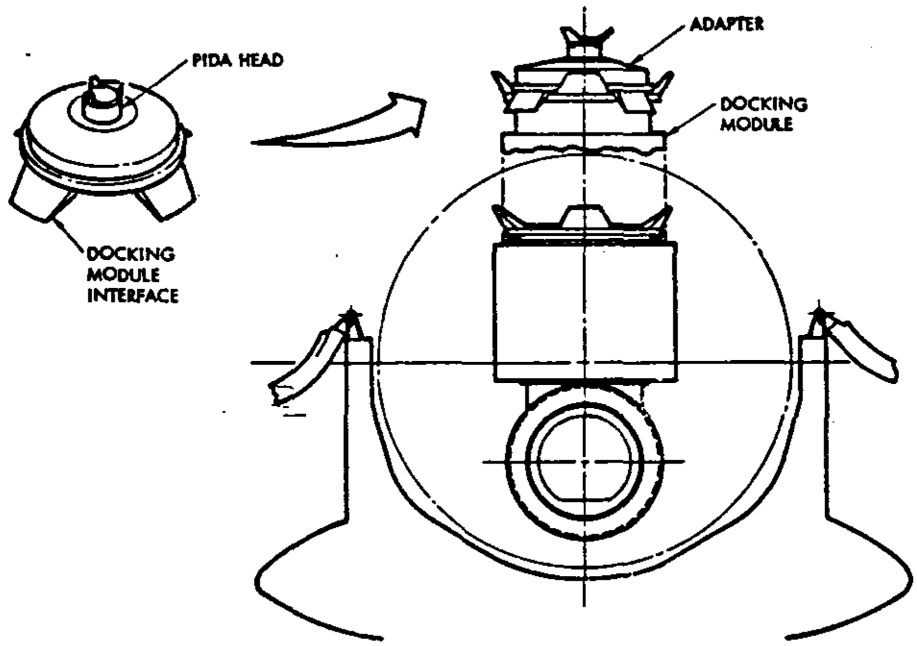

FIGURE 5.22 DOCKING MODULE ADAPTER

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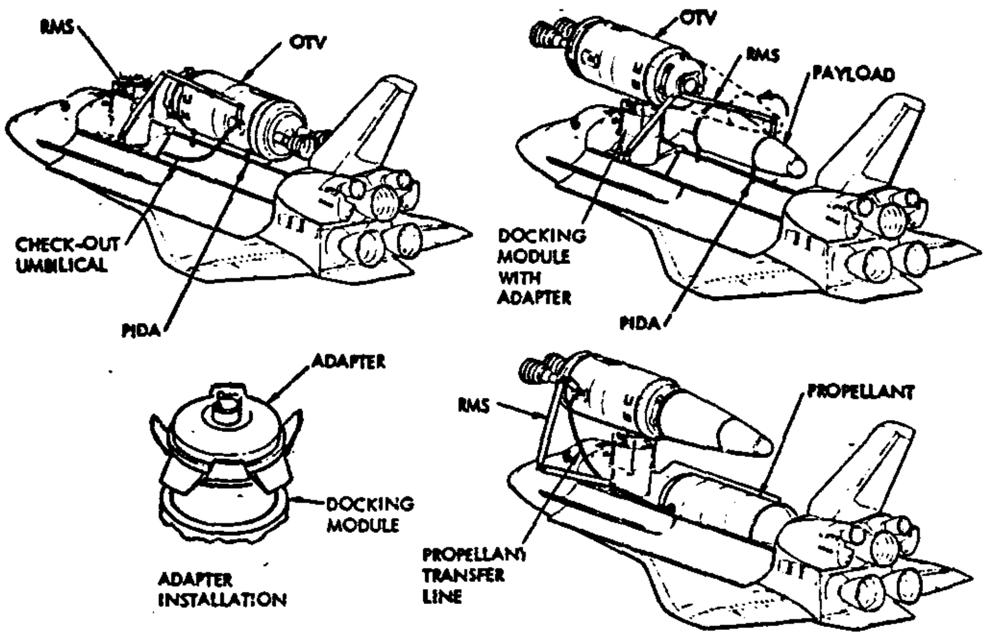

FIGURE 5.23 OTV/MOTV IN-ORBIT ASSEMBLY FROM ORBITER

No significant impacts to the orbiter to perform the OTV assembly, servicing, and checkout are anticipated. The docking/berthing adapter, propellant transfer equipment, and any unique checkout equipment are the only unique equipment identified to perform these functions.

REFERENCES

1. An Introduction to the Assurance of Human Performance in Space Systems, Martin Marietta Corp.

2. Manned Orbiral Transfer Vehicle (MOTV), Volume 5, Turnaround Analysis, Grumman Aerospace Corporation (November 7, 1979)

6.0 CONCLUSION

This section summarizes the conclusions that have resulted from the five basic tasks. The conclusions are organized by the SOC program elements rather than by tasks. The conclusion statements will emphasize the implications to the SOC, to the Shuttle, and to an OTV. The unique equipment required to support each of the elements is also included.

6.1 SOC IMPLICATIONS

- Fly a variable altitude to optimize Shuttle Logistics Delivery
- All module mating interfaces and other potential mating interfaces to be a "standard" configuration
- All interface ports to be "passive" - no active attenuation
- Provide appropriate location and number of RMS grappling points for all SOC modules and assemblies
- All modules to have two PIDA attach points for cargo bay deployment
- All hatches at berthing interface to contain a window and provisions for mounting a TV camera for berthing alignment viewing thru window
- Provide mounting for two lights within interface port for alignment target viewing
- Accommodate provisions on exterior of interface hatches for mounting alignment target
- Provide adequate CMG and RCS control to stabilize untended module assemblies during SOC buildup
- Primary Shuttle mating port to be oriented to permit orbiter tail down orientation
- Provide berthing/docking accommodations for second orbiter on habitable module connecting tunnel at second habitable volume
- SOC to provide fuel transfer control
- Accommodate fuel transfer line on SOC exterior including active orbiter interface segment
- Accommodate propellant storage facility (growth capability)

- Accommodate flight support facility
- Provide lights on RCM cab with tilt and pan capability
- Provide TV and a light on the RCM manipulator wrist and on the elbow both with tilt and pan capability
- Provide colored marker lights at extremities of SOC - approximately 45 required

6.2 SHUTTLE IMPLICATIONS

- Provide accommodations for a docking module interfacing with the Spacelab tunnel adapter
- RMS software modifications required for berthing orbiter to SOC
- Provide accommodations for two PIDA's
- Provide accommodations for a HPA
- Provide accommodations for two additional passenger seats in the mid deck of crew cabin
- Provide tilt and pan light on aft bulkhead
- Provide a HPA to standard mating interface adapter with a fixed light and TV kit

6.3 OTV IMPLICATIONS

- Provide standard berthing port at forward end of OTV and MOTV crew module
- Provide two PIDA attach points on body of OTV for Shuttle payload bay removal and servicing fixture attach
- Provide independent manned module for MOTV missions
- Provide externally mounted systems packages with two point attach provisions
- Provide electrical umbilical for servicing facility interface(s)
- Provide purging facility interface(s)

- Provide propellant fill interface(s)
- Provide engine removal/replacement capability (trade study)
- Provide external insulation removal/replacement capability (trade study)
- Provide other liquid and gas fill umbilical interfaces
- Provide grapple points for manipulator(s) and RMS

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